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Integrated Orbital Servicing Study for Low-Cost Payload Programs

MARTIN MARIETTA

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INTEGRATED ORBITAL SERVICING STUDY FOR LOW-COST PAYLOAD PROGRAMS

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FOREWORD

This study was performed under Contract NAS8-30820 for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under the direction of James R. Turner, the Contracting Officer's Representative. The final report consists of two volumes:

Volume I - Executive Summary,

Volume II - Technical and Cost Analysis.

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I. INTRODUCTION

Much of the Space Transportation System (STS) planning centered around the investigation of various operating methodologies to achieve low-cost space operations. Primary emphasis focused on justifying the STS development on an economic basis. The emphasis was to show that the development investment, initial fleet costs, and supporting facilities for the STS could be effectively offset by exploiting the capabilities of the STS to satisfy mission requirements and reduce the cost of payload programs. Although many items contribute to cost effective payload programs, the maintenance and/or refurbishment question, with its many variables, embraces a majority of the design, operation, and cost questions that must still be resolved before the full potential of the STS can be achieved.

Considerable work has already been done relative to the orbital maintenance question. The large number of maintenance studies performed for NASA and DOD over the past few years formed the basis for this study. These studies generally accented specific maintenance concepts, spacecraft programs, space tug effects, or certain analytical aspects. It was necessary to place all these alternative maintenance concepts on a common basis for effective comparison. This effort included an assessment of the relative value of the previously identified concepts and an overall comparison of the expendable, ground-refurbishable, and on-orbit maintainable modes. Through this process, the most effective concepts were isolated.

The following major conclusions were reached in the study.

- The development of an on-orbit servicer maintenance system is compatible with many spacecraft programs and is recommended as the most cost effective system.
- Spacecraft can be designed to be serviceable with acceptable design, weight, volume, and cost effects.
- Use of on-orbit servicing over the 12 years covered by the 1974 SSPD and the October 1973 Payload Model results in savings greater than
  - nine billion dollars over the expendable mode, and
  - four billion dollars over the ground refurbishable mode.

- The pivoting arm on-orbit servicer was selected and a preliminary design was prepared.
- Orbital maintenance does not have any significant impact on the space transportation system.
- Users need guarantees that servicing will be available and assurances that it will be cost effective.

The advantages of on-orbit servicing are greatest when there are many similar spacecraft in orbit, when the program time is long compared to the spacecraft lifetime, when the spacecraft availability requirement is similar for comparative modes, and when the spacecraft cost is not too low compared to the launch cost. The study outputs included a one-tenth scale mockup of the on-orbit servicer and three representative spacecraft as well as engineering test units of two forms--side- and bottom-mounting--of module interface mechanisms.

While the study used a NASA mission model representing automated spacecraft, the general conclusions are applicable to sortie missions and to DOD spacecraft. The study has been coordinated, integrated, and data exchanged with a parallel study, Integrated Orbital Servicing and Payloads Study, being conducted by the COMSAT Laboratories of the Communications Satellite Corporation (COMSAT) under the direction of Dr. Gary D. Gordon. The COMSAT study principally looked into on-orbit servicing and STS effects on communications satellite operations. These activities have been most beneficial to the conduct of this study.

A. STUDY OBJECTIVES

The broad objective of this integrated orbital servicing study (IOSS) was to provide the basis for the selection of a cost effective orbital maintenance system supported by the space transportation system. This objective required the selected mode to be cost effective in the sense of minimizing the total life-cycle spacecraft program costs, including those associated with maintenance, while retaining the spacecraft availability level implied by the payload model. The maintenance approach selected

could have been a combination of modes which could be selectively applied to the payload model automated spacecraft programs.

Inclusion of the study add-ons has expanded the objective to include preliminary design of a cost effective servicer, fabrication of a one-tenth scale mockup, evaluation of the control issues pertinent to servicing in orbit, expanded technical emphasis on spacecraft interfaces to better assess the potential effects of spacecraft configuration for servicing, and the design and fabrication of engineering test units of two different space-replaceable unit interface mechanisms and an associated end effector.

The large number of maintenance studies performed for NASA and DOD the past few years form the basis of this study as shown in Table I-1. These prior studies (Chapter XI) generally accented specific maintenance concepts, spacecraft programs, space tug effects, or certain analytical aspects. It was necessary to put the alternative maintenance concepts on a common basis for effective comparison. All cost-generating effects were to be identified so the cost comparison could be complete. The design effort was originally limited to "gap filling" as necessary to form a basis for generating costs.

Of the many approaches to providing servicing functions, module exchange was selected for maintenance concept evaluation because it satisfies the majority of the servicing operations with a single technique. This selection is consistent with the findings of the majority of the prior studies.

Table I-1 IOSS Scope

CONSIDERATIONS	ACTIVITIES
BUILD ON PRIOR STUDY RESULTS	PUT PRIOR WORK ON COMMON BASIS
INCLUDE ALL MAINTENANCE CONCEPTS	PERFORM TECHNICAL EVALUATIONS
ALL AUTOMATED SPACECRAFT IN PAYLOAD MODEL	CONDUCT STS IMPACT ANALYSIS
SHUTTLE ORBITER AND FULL CAPABILITY SPACE TUG	DETERMINE SPACECRAFT INTERFACE DESIGN REQUIREMENTS
PRIMARY SERVICING FUNCTION IS MODULE EXCHANGE	PERFORM CONSISTENT ECONOMIC ASSESSMENT
	EVALUATE PROGRAMMATIC/MANAGEMENT ASPECTS
	PREPARE SERVICER PRELIMINARY DESIGN AND MOCKUP
	IDENTIFY SERVICER CONTROL SYSTEM APPROACH
	DESIGN AND FABRICATE SRU INTERFACE MECHANISMS
	PREPARE STUDY RECOMMENDATIONS WITH SUPPORTING RATIONALE

Module exchange can provide the servicing functions of (1) repair failed equipment, (2) repair degraded equipment, (3) overcome design failures, (4) replace/replenish worn-out equipment, and (5) update equipment with new models. Equipment includes mission equipment as well as subsystem equipment. The maintenance concepts were also evaluated as to their adaptability to such other servicing functions as inspection, cleaning, and fault detection and isolation.

As the various maintenance concepts were identified, it became obvious that very little hard data existed; most concepts were just sketches of the spaceborne equipment and there were no data concerning the associated ground and operations equipment. Thus it was necessary to complete the concept definitions in many areas. Inherent in the activities of Table I-1 is identification of the criteria for the several evaluations. These criteria have been identified and evaluated and have become one of the significant study outputs.

In our examination of the many maintenance concepts, the entire automated spacecraft mission model, full life-cycle costs, the entire range of STS interfaces, and the myriad detail aspects, we found that the resultant breadth of our study permitted depth in only certain limited areas. We have compensated for this effect by drawing particularly on two excellent concurrent studies, Operations Analysis Study by the Aerospace Corporation, and Servicing the DSCS-II with the STS by TRW Systems Group. These studies concentrated on more limited aspects and provided the depth of analysis needed so we could apply it across the breadth of this study.

The automated spacecraft of the payload model were evaluated to identify those to which maintenance might reasonably be applied. This involved 47 different spacecraft programs with 340 missions. To provide the desired depth of analysis, six spacecraft programs were selected to characterize, or represent, all the maintenance-applicable spacecraft programs. The configurations of the six spacecraft in this characteristic set are shown in Figure I-1. The figure shows each spacecraft in its operating configuration to approximately the same scale for each spacecraft. The characteristic set spacecraft designations are biomedical experimental scientific satellite

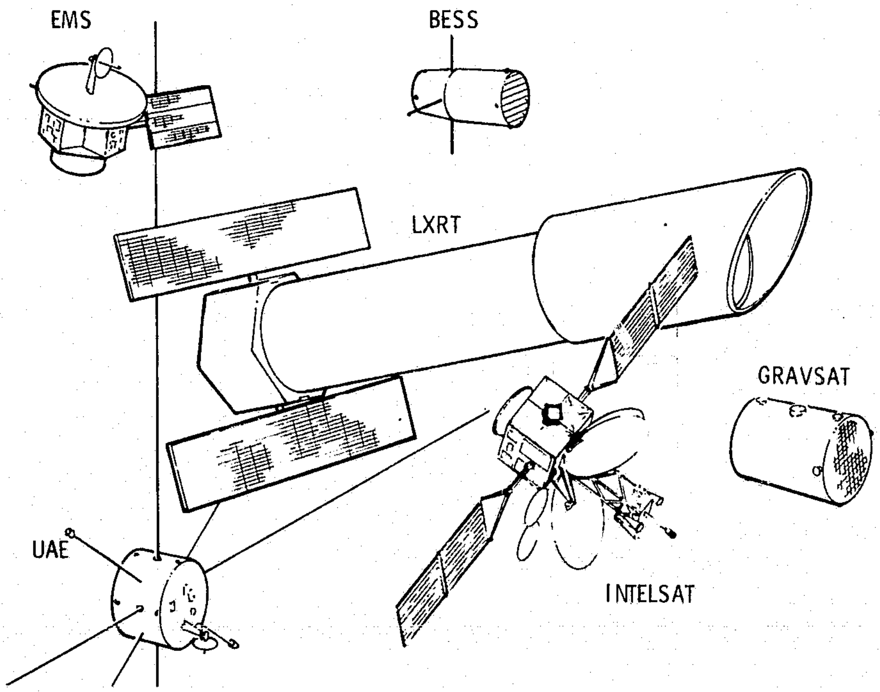


Figure I-1 Configurations of Characteristic Set

(BESS), environmental monitoring satellite (EMS), gravity satellite (GRAVSAT), international communications satellite (INTELSAT), large X-ray telescope (LXRT), and upper atmosphere explorer (UAE).

The figure illustrates the variety of shapes, sizes, and configurations of spacecraft that might be involved in servicing. The configurations of the spacecraft considered for maintenance are important for the following reasons:

1) The sizes and shapes of the spacecraft as stowed in the payload bay are necessary to calculate potential launch sharings and costs;

2) The operating configuration of the spacecraft as compared to the stowed configuration in the payload bay is necessary to determine requirements for reconfiguring the operating spacecraft to fit back into the payload bay for ground refurbishment;

3) The operating configuration is necessary for investigating docking considerations and movement of external servicing devices over the spacecraft surfaces; and

4) The current configuration is necessary to help determine if, and how, a spacecraft should be configured for servicing.

Figures I-2 and -3 illustrate serviceable configurations of the large X-ray telescope and the INTELSAT being serviced by an on-orbit servicer where the orbiter and tug are the respective carrier vehicles. These figures show two applications of the pivoting arm servicer, recommended by this study, that can also be applied to an earth-orbital teleoperator system, to a geosynchronous free-flyer, to the solar electric propulsion system, and to some forms of the interim upper stage.

B. RELATIONSHIP TO OTHER NASA EFFORTS

The IOSS, with its emphasis on building on prior and parallel study results, had a significant relationship to other NASA efforts. The prime relationship was with the Integrated Orbital Servicing and Payloads Study being performed by COMSAT Laboratories of the Communications Satellite Corporation. These two studies were conducted in parallel for the same MSFC Contracting Officer's Representative, James R. Turner. The studies were coordinated, integrated and data was exchanged. Monthly coordination meetings were held and all our formal presentations were joint. The purpose of the COMSAT effort is to include a commercial user's perspective and to provide a fuller consideration of the effects of servicing on INTELSAT design and operations.

The major part of the prior work, which included over three million dollars of contracted effort, is well represented by the seven studies of Table I-2. The recommendations from these studies and the types of data contained in the study reports are shown in Table I-3. These recommendations were useful because they provided a tentative set of conclusions the IOSS could support or reject. The study agrees with most of the stated recommendations as explained in Chapter II.

Table I-2 Significant Prior Studies

PAYLOAD SUPPORTING STUDIES FOR TUG ASSESSMENT MSFC IN-HOUSE, 1973
IN-SPACE SERVICING OF A DSP SATELLITE SAMSO/TRW, MARCH 1974
UNMANNED ORBITAL PLATFORM MSFC/RI, SEPTEMBER 1973
PAYLOAD UTILIZATION OF TUG MSFC/MDAC, GE AND FAIRCHILD, MAY 1974
OPERATIONS ANALYSIS NASA/AEROSPACE, JULY 1974
SERVICING THE DSCS-II WITH THE STS SAMSO/TRW, MARCH 1975
EARTH OBSERVATORY SATELLITE SYSTEM GSFC/IN-HOUSE AND CONTRACTED, CONTINUING

Two of the studies were particularly helpful. The operations analysis study by Aerospace defined

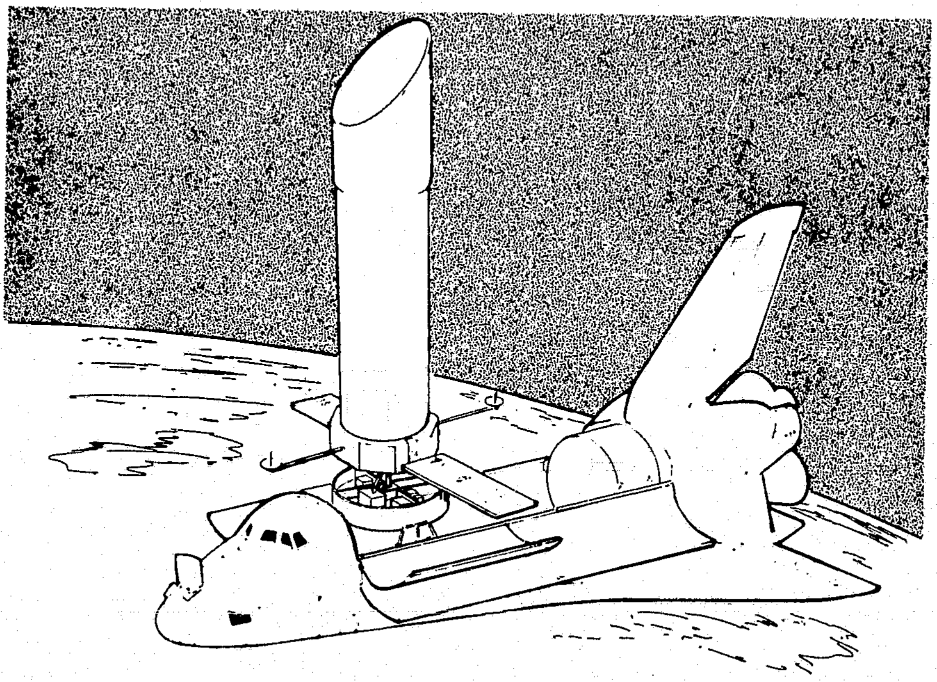


Figure I-2 Servicing the Large X-Ray Telescope at the Orbiter

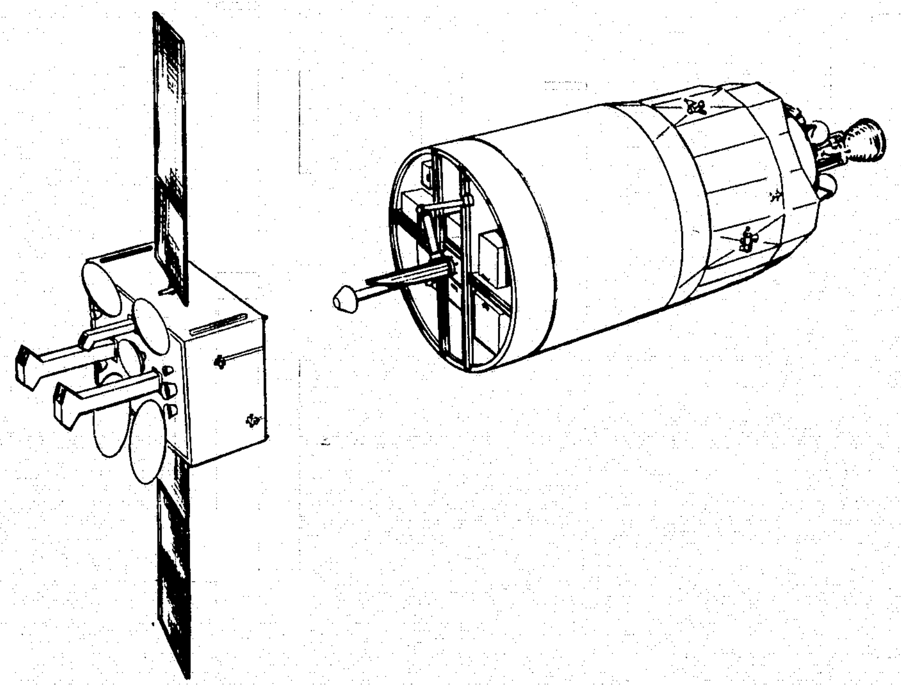


Figure I-3 Servicing the Intelsat via the Full-Capability Tug

Table I-3 Results of the Significant Prior Studies

THEIR RECOMMENDATIONS INCLUDED:
ON-ORBIT SERVICING IS THE MOST PROMISING MAINTENANCE APPROACH (ALL);
SPACECRAFT SHOULD BE DESIGNED FOR SERVICING (ALL);
GROUND REFURBISHMENT IS NOT AS PROMISING (SIX);
HIGH RELIABILITY MAY BE MORE COST EFFECTIVE (THREE);
ON-ORBIT SERVICING SHOULD BE FURTHER INVESTIGATED (ALL).
TYPES OF DATA AVAILABLE:
SERVICER CONCEPTS (ALL);
SERVICEABLE SPACECRAFT CONCEPTS (ALL);
COST DATA (ALL);
SERVICER EVALUATION CRITERIA (SIX);
RELIABILITY ASSESSMENT (FIVE);
MODULE SIZES AND WEIGHTS (SIX).

Table I-4 Concurrent Studies

MULTI-MISSION SUPPORT EQUIPMENT MSFC/MMC, JUNE 1974, 10 MONTHS
ORBITAL ASSEMBLY AND MAINTENANCE JSC/MMC, AUGUST 1974, 12 MONTHS
STUDY TO EVALUATE THE EFFECT OF EVA ON PAYLOAD SYSTEMS AMES/RI, JULY 1974, 6 MONTHS
MULTI-MISSION SUPPORT EQUIPMENT (LAUNCH SITE) MSFC/MMC, SEPTEMBER 1974, 8 MONTHS
EARTH ORBITAL TELEOPERATOR SYSTEM (EOTS) CONCEPTS AND ANALYSIS MSFC/MMC, JANUARY 1975, 12 MONTHS

a set of standardized modules and the complement of those modules for 29 spacecraft. It also provided weight and reliability data for these modules. The data were extrapolated to our set of 47 spacecraft programs. The DSCS-II study by TRW was based on existing TRW spacecraft and provided much detailed data on designs, costs, and schedule effects. These data helped us to extrapolate the NASA-provided spacecraft cost numbers from the expendable form to the ground-refurbishable and on-orbit serviceable forms of spacecraft.

The statement "high reliability may be more cost effective" can be interpreted in two ways. Two of the studies concluded that high reliability may be more cost effective than orbital servicing, while the third study concluded that orbital servicing is more cost effective than the other two modes and, within this mode, the reliability increases considered provided additional savings for the spacecraft system considered.

Table I-4 lists five concurrent studies that provided additional data helpful to the IOSS and to which the IOSS provided significant and useful data.

C. STUDY APPROACH

The objective of maintenance is to increase a system's availability, which is a measure of the time that a system is ready to perform its intended mission. Maintenance, or servicing, is one way to reduce the cost of availability. The many approaches to obtaining spacecraft availability are shown in Figure I-4. This tree of approaches is easily divided into

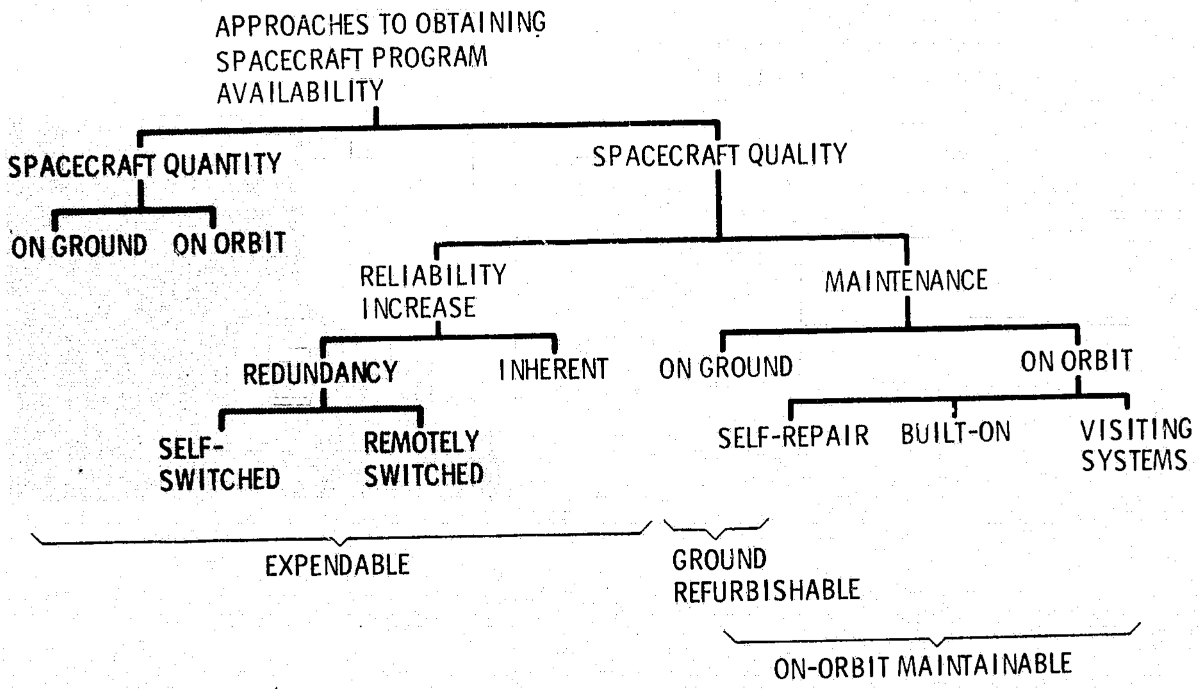


Figure I-4 Spacecraft Program Availability Approaches

the maintenance modes of the study--expendable, ground-refurbishable, and on-orbit maintainable. Two maintenance concepts shown were considered and found to have little application--built-on and self-repair. Built-on is a maintenance concept in which the spacecraft has its own spare modules and the failed modules are replaced mechanically. Self-repair is an extension of built-on where the spacecraft has a second manipulator that is used to repair the failed modules. Note that the availability approaches in the shaded area are not considered part of the study effort; those in the unshaded area were addressed.

In the expendable mode, spacecraft are launched until the desired on-orbit fleet size is obtained and then each failed spacecraft is replaced

with a new spacecraft. The ground-refurbishable mode starts as with the expendable mode until a spacecraft fails. Then the failed spacecraft is returned to earth, repaired, and relaunched. (If an extra spacecraft has been procured, then it is sometimes possible to launch the replacement spacecraft and retrieve the failed spacecraft on one mission.) The on-orbit maintainable mode is also like the expendable mode until a failure occurs. Then replacement modules are taken into space, exchanged with the failed modules, and the spacecraft returned to normal operation. The method used for exchanging the modules, called visiting systems, has been the subject of much study.

The overall study task identification and interrelationships shown in Figure I-5 demonstrate the highly interactive approach necessary for the technical and economic evaluations to support the study objective--provide the basis for selection of a cost effective orbital maintenance system supported by the STS. The desired results are tradeoff studies, rationale, evaluations, criteria, spacecraft configuration data, and cost structure formats to support the selection of maintenance concepts to be used in the actual cost determination of on-orbit serviceable versus expendable and ground-refurbishable alternatives that will provide the desired spacecraft availability.

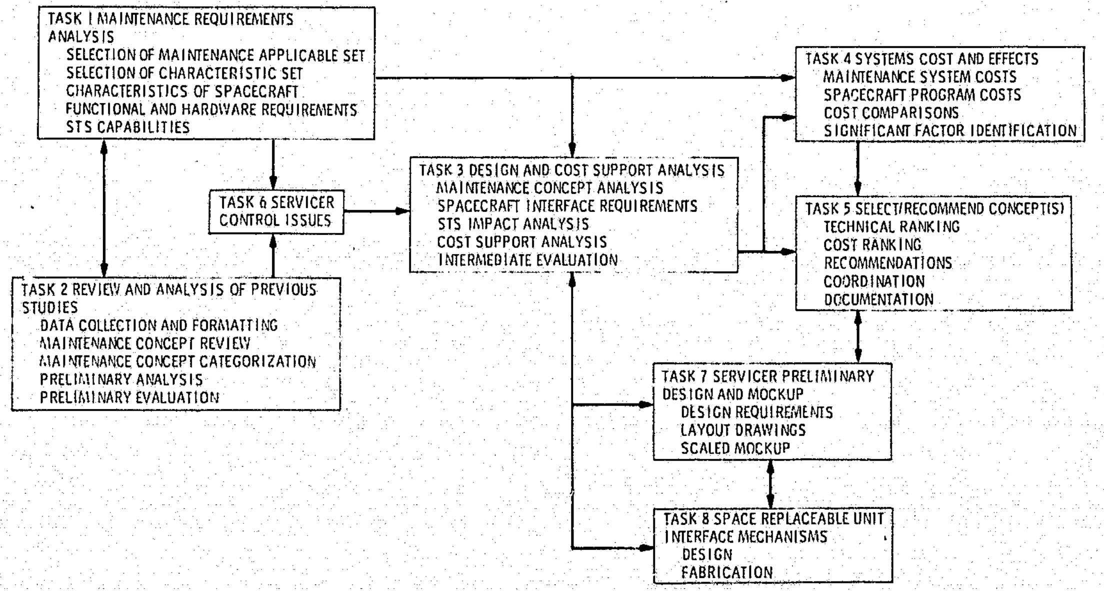


Figure I-5 Study Tasks and Flow

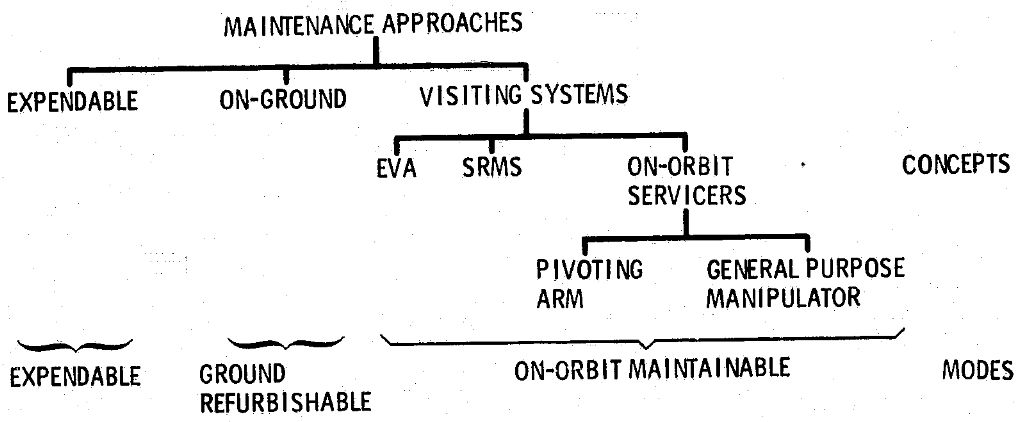
After the first quarterly review the study effort was increased to include an increment to task 3, spacecraft interface requirements; task 6, servicer control issues; and task 7, servicer preliminary design and mockup. These activities were added to meet the important needs of providing an effective on-orbit servicing demonstration device; i.e., a servicer mockup, an initial evaluation of the controls problem, and expanded definition of the effect on the spacecraft interfaces resulting from the on-orbit servicing scenario. After the third quarterly review, task 8, space-replaceable unit interface mechanism design and fabrication, was added to provide engineering test units for two approaches to the important mechanical fastening interface between the space-replaceable units and the spacecraft and stowage rack.

This technical volume has been organized to introduce each subject in sequence and to complete the discussion of that subject in its own chapter. The result is a different grouping of subjects from the study tasks. Table I-5 shows, on the left hand side, which chapters discuss the subjects of specific tasks. The chapters have been ordered by completeness of discussion with the primary chapters listed first. The inverse correspondence is shown on the right hand side of the table.

Table I-5 Task/Chapter Correspondence

TASK TO CHAPTER		CHAPTER TO TASK	
TASK	CHAPTERS	CHAPTER	TASKS
1	III	I	--
2	IV, II	II	ALL, 2
3	IV, V, VIII, IX	III	1
4	IX, VII, VIII, V	IV	2, 3, 5
5	X, IV, V, IX	V	3, 8, 4, 5
6	VI	VI	6
7	VII	VII	7, 4
8	V	VIII	3, 4
		IX	4, 5, 3
		X	5

While the study involved many facets, iterations, and evaluations, the flow of the major tradeoffs is shown in Figure I-6. The three *modes*



EVALUATION CONSIDERATIONS:

1. SPACECRAFT DESIGN ASPECTS
2. STS IMPACTS
3. TECHNICAL
4. OPERATIONAL AREAS
5. PROGRAMMATIC
6. COST

Figure I-6 Study Evaluation Flow

are shown across the middle of the figure. Each *mode* can be achieved by one or more maintenance *concepts*. The primary tradeoff is between the three *modes*, but the maintenance *concept* for the visiting system level--EVA versus shuttle remote manipulator system vs on-orbit servicer--also had to be selected. While Figure I-6 expresses the on-orbit servicer tradeoff as being between the pivoting arm and the general-purpose manipulator maintenance *concepts*, these two concepts are the result of a screening/categorization/evaluation process that started with 15 different concepts. The considerations shown on the figure were used in the evaluations depicted as well as in most of the other study evaluations.

D. SIGNIFICANT CONCLUSIONS AND RESULTS

The significant conclusions and results reached in the two Integrated Orbital Servicing studies are presented below with the major conclusions shown in italics. Many secondary results and supporting conclusions are given in the rest of this chapter and in the technical volume. The following significant conclusions and results were generated by both COMSAT and

Martin Marietta in their two companion studies. These conclusions, where Martin Marietta has performed a significant part of the work, are discussed and their supporting rationale are presented in the remainder of this chapter.

1. <u>Top-level conclusions</u>

   a) *On-orbit maintenance is the most cost-effective mode* (Chapter X).
   b) *Spacecraft can be designed to be serviceable with acceptable design, weight, volume, and cost effects* (Chapter V).
   c) The module exchange form of servicing is applicable to repairing failed satellites, improving reliability of operating satellites, and updating equipment (Chapter II).
   d) Analysis, design, engineering test unit fabrication, and evaluation of on-orbit servicers should continue (Chapter VII).
   e) On-orbit servicing can increase program flexibility and satellite reliability, lifetime, and availability (Chapter II).
   f) Ground refurbishment is not cost effective for most geosynchronous satellites (Chapter IX).

2. <u>Maintenance concepts</u>

   a) *The on-orbit servicer maintenance concept is recommended* (Chapter X).
   b) The on-orbit servicer, extravehicular activity, and shuttle remote manipulator system are all technically feasible (Chapter IV).
   c) Only the on-orbit servicer is applicable to both tug and orbiter based missions (Chapter IV).
   d) Remote control of module exchange with an on-orbit servicer is technically feasible (Chapter VI).

3. <u>On-orbit servicers</u>

   a) *The pivoting arm on-orbit servicer was selected and a preliminary design was prepared* (Chapters IV and VII).
   b) On-orbit servicer concepts exist that will permit a broad range of spacecraft design alternatives (Chapter IV).
   c) On-orbit servicing is compatible with standardized modules or spacecraft, but does not require them to be cost effective (Chapter V).

d) Side- and bottom-mounting forms of space replaceable unit interface mechanisms are useful and have been designed (Chapter V).

4. Economics evaluations

a) *Use of on-orbit servicing over the 12 years covered by the 1974 SSPD and the October 1973 Payload Model results in savings greater than*

- *nine billion dollars over the expendable mode, and*
- *four billion dollars over the ground refurbishable mode*

(Chapter IX).

b) The life cycle costs of the on-orbit servicer represent approximately one percent of the overall savings and these costs can be fully recovered by 1982 (Chapter IX).

c) Cost sensitivity analyses showed that wide variations in cost data, especially mission model size and fraction of spacecraft replaced, affect specific savings but do not change the major study conclusions (Chapter IX).

d) A long-life free-flying servicer at geostationary orbit is potentially cost effective (Chapter IX).

e) Specific launch cost reimbursement policies can be an important factor in which form of servicing is adopted for individual spacecraft programs (Chapter IX).

f) Expendable satellites are cost effective where satellite lifetime meets program lifetime requirements (Chapter IX).

5. Development implications

a) *A single development of an on-orbit servicer maintenance system is compatible with many spacecraft programs and is recommended* (Chapter V).

b) *Orbital maintenance does not have any significant impact on the space transportation system* (Chapter VIII).

c) On-orbit maintenance with the pivoting arm servicer is compatible with a variety of delivery vehicles such as the orbiter, full capability tug, free-flying servicer, solar electric propulsion system, earth orbital teleoperator system, and some forms of the interim upper stage (Chapter IV).

6. User acceptance

a) *Users need guarantees that servicing will be available and assurances that it will be cost effective.*

b) A deeper understanding of the orbital servicing cost structure is required before initiating drastic changes in conventional satellite construction and operations methods.

c) Scheduling delays of several months are tolerable for many servicing requirements.

d) Development of the on-orbit servicer should include early in-space demonstrations of module exchange along with rendezvous and docking (Chapter X).

e) Building, flying, and servicing a serviceable satellite is needed to obtain widespread acceptance of orbital servicing (Chapter X).

E. IMPLICATIONS FOR RESEARCH

All the on-orbit servicers considered--especially the one recommended--used approaches, components, techniques, and arrangements that are well within present day state of the art. However, several associated aspects have been identified as candidate supporting research and technology items in the advanced development category. These are discussed in the following paragraphs.

1. Control Techniques for On-Orbit Servicers (Chapter VI)

This study recommended a combination of supervisory and remotely manned control. These techniques should be further considered to ensure that the most effective system of control of the module exchange process is employed.

2. Space-Replaceable Unit Interface Mechanisms (Chapter V)

The mechanical interface between space-replaceable units and the spacecraft and stowage rack needs a level of standardization if a single servicing concept is to be used across many spacecraft programs. Although two versions of the SRU interface mechanism have been designed and engineering test units fabricated, a significant amount of technology and development work must be performed before any interface mechanism can be established as a standard.

3. <u>Connectors</u> (Chapter V)

When modules or SRUs are exchanged, connectors will be demated and mated with a single push-or-pull action. No such connectors suitable to this use were found and they must be developed. In addition to the usual electrical power and electronic signal connectors, waveguide connectors are needed. There is also a probable need for fluid connectors and some consideration should be given to thermal connectors.

4. <u>On-Orbit Servicing One-G Demonstration Facility</u> (Chapter VII)

This facility is needed to study the exchange of modules in one-g so control systems, latches, trajectories, connectors, and tolerances can be investigated and basic data developed for application to flight hardware development.

5. <u>Long-Term Space Environmental Effects</u> (Chapter V)

The long-term effects of the space environment on the ability to replace modules and on continued operation of the various parts of the nonreplaceable units are not known. It is desirable to verify predictions that modules can be replaced and that the nonreplaceable units will have an adequately long life.

6. <u>Contamination Protection</u> (Chapter V)

The contamination limits for spacecraft during on-orbit servicing should be established so the appropriate limits for the on-or-it servicer and its carrier vehicle can be established. The servicer itself and the stowage rack can be kept clean by proper shrouding if necessary. However, the carrier vehicles, i.e., orbiter and tug, are not so easily kept clean and development of a "clean" earth orbital teleoperator system should be considered if contamination limits are too stringent.

7. <u>Space-Replaceable Solar Arrays and Drives</u> (Chapter V)

Solar arrays and drives are expensive items that were considered as part of the nonreplaceable units but that possibly should be considered for development into space-replaceable units.

F. SUGGESTED ADDITIONAL REPORT

A review of the IOSS efforts and conclusions identified a number of areas that merit consideration for substantial additional effort. They are as follows.

1. Engineering aspects,

   a) Analysis, design, engineering test unit fabrication, and evaluation of on-orbit servicers (Conclusion 1-d),

   b) Development of SRU interface mechanisms (Conclusion 1-b),

   c) Development of electrical, waveguide, and fluid connectors compatible with SRU interface mechanisms (Conclusion 1-b),

   d) Simulations of module exchange including full-scale SRU interface mechanisms (Conclusion 3-a),

   e) Investigation of on-orbit servicer control following the approach that has been suggested (Conclusion 2-e),

   f) Design of representative serviceable spacecraft (Conclusion 1-b),

   g) Development of spacecraft structural configurations that are compatible with space-replaceable units (Conclusions 1-b),

   h) Investigation of multiple payload rendezvous techniques and energy requirements (Conclusion 4-b),

   i) Evaluation of need for, and possible development of, a thermal connector (Conclusion 1-b),

   j) Investigation of alternative materials in on-orbit servicer designs (Conclusion 3-a),

2. Economic aspects,

   a) Development of better cost data including spacecraft standardization, flight density, and scheduling effects (Conclusion 4-a),

   b) Generation of confidence limits on cost data (Conclusion 4-a),

   c) Application to DOD programs (Study Limitations of Vol. I),

   d) Investigation of potential servicer benefits with other spacecraft not in the mission model considered herein; i.e., sortie lab payloads, planetary, lunar, and heliocentric spacecraft (Study Limitations of Vol. I),

e) Determination of effects of the continuing development of NASA launch cost reimbursement policy plans on economics and operations of servicing (Conclusion 4-e),

f) Investigation of availability, lifetime, and servicing strategies with a reliability simulation (Conclusion 1-e),

3. Management aspects (Conclusion 2-a),

a) Development of on-orbit servicer implementation plan,

b) Investigation of programmatic/scheduling aspects of the STS,

c) Consideration of operational mode alternatives,

d) Evaluation of compatibility of interim upper stage with on-orbit servicing,

e) Consideration of orbit-based servicers (chemical vs solar electric propulsion),

f) Development of techniques for spacecraft program manager selection of maintenance modes,

g) Identification of safety implications,

h) Evaluation of adaptability of the on-orbit servicer to central or peripheral docking systems;

4. User aspects (Conclusion 5-a),

a) Development of an on-orbit servicer demonstration plan including on-orbit demonstrations,

b) Identification and fabrication of equipment for concept verification and test facility.

G. STUDY GUIDELINES

The following guidelines were abstracted from the study Request for Proposal for reference purposes.

1) This study will make maximum usage of the previous orbital maintenance conceptual studies and/or analyses. All methods are to be compared; therefore, no effort is being made in the beginning to restrict the range of consideration.

2) Only automated spacecraft are to be considered. In this context automated refers to spacecraft which are delivered to orbit and released and operate without the direct presence of man.

3) The integrated orbital servicing system shall be compatible with the shuttle and the high technology (full capability) tug.

4) This study shall be applicable to the current NASA traffic and mission models existing during the period of study.

5) The payloads/spacecraft descriptions to be utilized in this study shall be defined by the Shuttle Systems Payload Data (SSPD) existing during the contract period.

II. PRIOR SERVICING STUDIES

The background of this study is found in the large number of orbital maintenance studies which were completed or in progress when the IOSS started. One of the IOSS conditions is that it use this prior work as a starting point. It is thus useful to discuss these prior studies and their application to IOSS. They included many different aspects of spacecraft servicing, each had their own specific objectives, approaches, and areas of interest. Most tended to have positive conclusions on the value of servicing. Some concluded that servicing was marginal eonomically but that the advantages gained would be worth the development. Others, of which the Institute for Defense Analyses paper, Current (FY 73) Issues Regarding Reusability of Spacecraft and Upper Stages for Military Missions (Chapter XI, Item 0-5) is representative; concluded that on-orbit servicing was marginally better economically but that the programmatic disadvantages and development problems militated against it.

The majority of the prior work may be loosely grouped into seven continuing activities whose current conclusions are represented by the studies listed in Table II-1. The symbols in parentheses after the study name are used as designators on other figures in this chapter. Specific references

Table II-1 Significant Prior Studies

PAYLOAD SUPPORTING STUDIES FOR TUG ASSESSMENT MSFC IN-HOUSE, 1973 (TA)
IN-SPACE SERVICING OF A DSP SATELLITE SAMSO/TRW, MARCH 1974 (DSP)
UNMANNED ORBITAL PLATFORM MSFC/RI, SEPTEMBER 1973 (UOP)
PAYLOAD UTILIZATION OF TUG MSFC/MDAC, GE AND FAIRCHILD, MAY 1974 (PUT)
OPERATIONS ANALYSIS NASA/AEROSPACE, JULY 1974 (OA)
SERVICING THE DSCS-II WITH THE STS SAMSO/TRW, MARCH 1975 (DSCS II)
EARTH OBSERVATORY SATELLITE SYSTEM GSFC/IN-HOUSE AND CONTRACTED, CONTINUING (EOS)

to the work may be found in the Bibliography of Chapter XI. The data and results of the prior studies, summarized in Table II-2, were used to identify directions and provide data and methodology for this study. The various items were analyzed, cross-checked between studies, and verified before being incorporated in the IOSS conclusions.

Table II-2 Results of the Significant Prior Studies

THEIR RECOMMENDATIONS INCLUDED:

- ON-ORBIT SERVICING IS THE MOST PROMISING MAINTENANCE APPROACH (ALL);
- SPACECRAFT SHOULD BE DESIGNED FOR SERVICING (ALL);
- GROUND REFURBISHMENT IS NOT AS PROMISING (SIX);
- HIGH RELIABILITY MAY BE MORE COST EFFECTIVE (THREE);
- ON-ORBIT SERVICING SHOULD BE FURTHER INVESTIGATED (ALL).

TYPES OF DATA AVAILABLE:

- SERVICER CONCEPTS (ALL);
- SERVICEABLE SPACECRAFT CONCEPTS (ALL);
- COST DATA (ALL);
- SERVICER EVALUATION CRITERIA (SIX);
- RELIABILITY ASSESSMENT (FIVE);
- MODULE SIZES AND WEIGHTS (SIX).

A. SUMMARY OF RECOMMENDATIONS

All the seven studies addressed on-orbit servicing as a significant STS capability. The questions were in the nature of how and under what conditions. The impetus to evaluate on-orbit servicing was provided by the activity to justify the space transportation system. The momentum was increased by the series of low cost payload studies performed by Lockheed Missiles and Space Company for NASA. This was followed by a series of point designs such as the earth observatory satellite, large space telescope, and defense support program. However, every study was limited to one or a few spacecraft or to specific aspects of on-orbit servicing.

The general, servicing-related, recommendations of the seven studies are given in Table II-2. Each study made positive statements on the

value of on-orbit maintenance and its superiority to the ground refurbishable maintenance mode and the expendable mode. The relative benefit of the ground refurbishable and expendable modes was not as clear cut. The operations analysis study did not comment on ground refurbishment and four of the studies did not comment on the relative value of high reliability. The tug assessment study did comment that long life, obtained by high reliability and reduction of wear-out aspects, could be cost effective. None of the other studies commented to this point. The IOSS has agreed with the first four recommendations of Table II-2 but has done no work to support or refute the fifth recommendation.

B. SCOPE AND DATA AVAILABLE

The scope of the seven prior studies with respect to servicing is shown in Table II-3. The emphasis and type of servicing data is given for each study. Most of the studies used reliability simulations to obtain numbers of spacecraft failures that could be used in the costing analyses. In some cases the reliability simulations were used with a costing model.

The types of data given in Table II-4 are those that could be used by the IOSS study and which were available in at least one of the studies. The operations analysis (OA) study by Aerospace Corporation is the most inclusive and applicable by subject and the DSCS-II study by TRW Systems Group went into the greatest depth with regard to spacecraft design and cost/reliability simulations. While these analyses and data were of the type we needed, they were used with care and with a full realization of the objectives, ground rules, and limitations of the reference work.

The reliability simulations provided some data on the relationships between availability, servicing strategies, and cost. Using the DSCS-II study as the primary basis, several points may be stated. High availability (>0.97) can best be obtained with the use of operating on-orbit spares. The major down-time contributor will then be the time to phase the spare spacecraft to its operating location. Ground preparations and orbiter scheduling inhibit obtaining high availability by spares on the ground. When on-orbit spares are used, then on-orbit servicing can reduce costs while maintaining the minimum availability at a high level. The number of maintenance missions (whether to replace or service spacecraft) tends to be small. For example, a ten year program with four

Table II-3 Scope of Prior Studies

TA	- DOMSAT B SPACECRAFT - LIMITED CONSIDERATION OF SERVICER OPTIONS - EMPHASIS ON HIGH AVAILABILITY, MAINTENANCE POLICIES, AND UPPER STAGE ALTERNATIVES
DSP	- DSP SPACECRAFT - LIMITED CONSIDERATION OF SERVICER OPTIONS - AVAILABILITY REQUIREMENT NOT CONSTRAINED
UOP	- LIMITED CONSIDERATION OF SERVICER OPTIONS - NO RELIABILITY SIMULATION - CAPTURED 60-80% OF MAINTENANCE APPLICABLE MISSIONS
PUT	- FOUR GEOSYNCHRONOUS SPACECRAFT - EMPHASIS ON EXPLOITING THE STS
OA	- 29 PROGRAMS REPRESENTED THE 42 MAINTENANCE APPLICABLE SPACECRAFT - LIMITED CONSIDERATION OF SERVICER OPTIONS - NO CONSIDERATION OF GROUND REFURBISHMENT - EMPHASIS ON UPPER STAGE ALTERNATIVES
DSCSII	- DSCSII SPACECRAFT - INCLUDES MAINTENANCE POLICIES AND UPPER STAGE ALTERNATIVES - INCLUDES SPACECRAFT EQUIPMENT FAILURE CLASSIFICATION - DETAIL SPACECRAFT AND SERVICER DATA
EOS	- EOS SPACECRAFT - LIMITED CONSIDERATION OF SERVICER OPTIONS - AVAILABILITY REQUIREMENT NOT CONSTRAINED - DETAIL SPACECRAFT AND SERVICER DATA

operating spacecraft might require four maintenance missions on the average. Each servicing mission replaces 15% of the modules on each of two spacecraft visited on the average. One result of there being few maintenance missions is that the cost per mission of keeping a cadre of knowledgeable people available (for the period from end of production to end of program) becomes high. As different sets of people are required for each spacecraft program, there is no proportionate savings for multiple spacecraft programs.

Table II-4 Types of Data Available From Prior Studies

	TA	DSP	UOP	PUT	OA	DSCSII	EOS
SERVICER EVALUATION CRITERIA	X		X	X	X	X	
SERVICER CONCEPT	X	X	X	X	X	X	X
SERVICEABLE SPACECRAFT CONCEPTS	X	X	X	X	X	X	X
MODULE SIZES AND WEIGHTS		X	X		X	X	X
MISSION EQUIPMENT MODULES			X		X		X
RELIABILITY ASSESSMENT	X	X	X		X	X	X
RELIABILITY SIMULATION MODEL	X	X			X	X	X
COST DATA	X	X	X	X	X	X	X
COSTING METHODOLOGY	X		X			X	
COST SENSITIVITY ANALYSIS	X			X			
FUNCTIONAL FLOW DIAGRAMS					X		
SOFTWARE COST ESTIMATES					X		

C. FUNCTIONAL AND DESIGN GUIDELINES

Functional and design guidelines for orbital maintenance were also abstracted from the prior studies. The functional guidelines are shown in Table II-5. Note the low level of consensus on specific functional guidelines. On only one item are all seven studies agreed, although more than one study recommended similarly on seven of the ten items.

These functional guidelines and the design guidelines were used as possible guidelines in the IOSS study. They were further evaluated as the study progressed and some were included in the study recommendations. The first item has been used as an IOSS guideline. It appears advantageous to use similar equipment for high earth orbits (HEO) and low earth orbits (LEO) if technically feasible so that the development cost of a second system can be avoided. The third item is involved in a national policy question as to whether anything will be left in space. The IOSS recommended that modules and spacecraft be left in space, for some cases, to save cost. However, if policy becomes one of reducing pollution in space by retrieving all space debris, then the cost of expendable spacecraft will increase and on-orbit maintenance will appear even better. The availability

Table II-5 Candidate Functional Guidelines From Prior Studies

LIMIT SERVICING TO MODULE EXCHANGE (ALL)
SIMILAR EQUIPMENT FOR LEO AND HEO (UOP, OA)
RETURN FAILED MODULES (UOP, OA, EOS, DSP SAID NO)
REPLACE LIFE LIMITED EQUIPMENT AND PROPULSION MODULES WHEN REPLACING FAILED MODULE (DSCSII, EOS)
AVAILABILITY REQUIREMENT $\approx$ 0.9999 (TA)
REPAIR COST FRACTION = 0.3 (TA)
WARNING INITIATED SERVICING PROGRAMS COST MORE (DSP, DSCSII)
ONE EXTRA FLEET SIZE IS GENERALLY BEST (TA, DSCSII)
MANNED SUPPORT FOR CONTINGENCIES (OA, EOS)
A PRE-SHUTTLE DEMONSTRATION PROGRAM IS REQUIRED (OA)

requirement of the TA study is more stringent than the availability numbers discussed in other studies where availability was used as a reliability simulation output rather than as a requirement. The one extra fleet size means that one more spacecraft is purchased and flown on-orbit than is required to meet the scheduled on-orbit requirement.

Every one of the studies recommended that on-orbit servicing be limited to module exchange. Module exchange can accomplish:

1) Repair of failures,
2) Repair of degraded equipment to improve spacecraft reliability,
3) Repair of design failures,
4) Repair of worn-out equipment and replenishment of expendables such as propellants, and
5) Updating of equipment in terms of more modern equipment or to change the mission objective.

Other functions such as cleaning, inspection, and checkout were not given significant importance as on-orbit servicing activities. There is part of a serviceable spacecraft called the nonreplaceable unit (NRU) which normally is not exchanged. It generally consists of structure, antennas, wiring, and solar arrays. All of these have a high reliability and thus represent few failures. The DSCS-II showed that on the average 0.25

spacecraft would be lost in attempting to operate four spacecraft over a ten-year program due to NRU failures. The result is that the large majority of servicing requirements can be met by the single activity of exchanging modules. To add more servicing functions will require significantly more servicer capability. Note that the spacecraft can be designed to do its own fault detection and checkout. The IOSS has thus concentrated on the important servicing activity of module exchange.

The design guidelines are shown in Table II-6. While it was easier to identify design guidelines than functional guidelines, the consensus

Table II-6 Candidate Design Guidelines From Prior Studies

USE DATA BUS CONCEPT (ALL)
SERVICER/SPACECRAFT INTERFACE MUST BE STANDARDIZED (PUT, EOS)
FLAT MODULE MOUNTING SURFACE (UOP, TA)
SIDE MODULE MOUNTING (DSCSII, OA)
SERVICER LENGTH $\leq$ 3 FEET (PUT)
SERVICER WEIGHT $\leq$ 500 LB (PUT)
AVOID CONDUCTIVE HEAT TRANSFER (OA, DSP)
SMALL REPLACEABLE MODULES ARE PREFERRED (TA)
SRU SIZE HAS MINOR EFFECT ON NUMBER OF SERVICING FLIGHTS (DSCSII)
SMALL SRUs REQUIRE HALF THE PER FLIGHT REPLACEMENT WEIGHT (DSCSII)
CARRY $\approx$ 1500 LB OF MODULES PER SERVICE MISSION (DSP)
WEIGHT AND COST INCREMENTS FOR ON-ORBIT SERVICING ARE $\approx$ 10% (TA)
WEIGHT PENALTY FOR SERVICEABILITY $\approx$ 1700 LB (OA), 400 LB (DSCSII)
SOFTWARE COSTS NOT EXCESSIVE (OA)

on recommendations is no better. Acceptance of the data bus concept for the spacecraft is almost mandatory for on-orbit maintenance, otherwise the number of pins in the mating electrical connectors will become excessive. This is particularly true if functional redundancy through the electrical connectors is used. The DSCS-II study was unique in that it involved three sizes of modules. The medium size module (15 per spacecraft) resulted in the lowest total program cost. The larger size modules resulted in a larger total module weight to orbit for the whole program while the

converse was true for the smaller modules. This can imply a larger launch cost for large modules depending on the launch cost reimbursement policy used. The DSCS-II study recommended use of large (eight per spacecraft) modules. The EOS study used four subsystem modules and six mission modules. The IOSS suggests the use of between twelve and twenty modules per spacecraft.

The PUT study length and weight limits were established by the orbiter cargo bay length constraint when carrying the tug and by the tug round trip weight capability. The three candidate guidelines on spacecraft geometry (items 2, 3, and 4) simplify the module exchange mechanism design but could cause the spacecraft design to be more difficult. Side mounting of modules and bottom (flat mounting surface) mounting of modules were both suggested. Note also the differences in spacecraft weight penalties incurred when the spacecraft is designed for on-orbit servicing. The TA number was for a single spacecraft and the OA number was averaged over 16 spacecraft. The OA number was large because almost every piece of mission equipment was individually modularized. Also the OA study involved an additional 121 lbs associated with each module which was due to: baseplate, mechanism, thermal control, electrical connector, and power conditioning. The DSCS-II raw data showed an additional 800 lbs of propellant to allow for stationkeeping and inclination control over the ten year program life. The IOSS suggests that 600 to 800 lbs per spacecraft is a more likely penalty for serviceable spacecraft design.

D. CONDITIONS FAVORING MAINTENANCE

Documentation for the seven studies was reviewed to isolate those spacecraft program/servicing conditions that might favor orbital servicing. These conditions are listed in Table II-7.

There are only two areas where the prior studies agree as to what favors orbital maintenance. These are launch cost sharing and multiple spacecraft servicing, both of which reduce the effective launch costs of a service mission. In no other area is a consensus obtained. This illustrates the difficulty of reaching simple, definite conclusions on orbital maintenance.

It had earlier been suggested that high spacecraft cost should favor the application of maintenance. However, none of the referenced studies

Table II-7 Conditions Favoring Orbital Maintenance as Identified By Prior Studies

	TA	DSP	UOP	PUT	OA	DSCS II	EOS
LAUNCH COST SHARING	X	X		X	X		
MULTIPLE SPACECRAFT SERVICING		X		X	X	X	
LOWER SERVICER COSTS	X			X			X
MORE SPACECRAFT PER PROGRAM				X			
SUBSYSTEM STANDARDIZATION			X				X
LOWER SPACECRAFT LIFETIME	X	NO		X			
LOWER RELIABILITY	NO			X		X	
GREATER UPPER STAGE PERFORMANCE	NO				X	NO	
USE OF SEPS					X		

identified this factor as favoring maintenance. The IOSS has concluded that the advantages of on-orbit servicing are greatest when there are many similar spacecraft in orbit, when the program time is long compared to the spacecraft mean time-to-failure, when the launch charges are based on weight or volume taken to and from orbit, when the spacecraft availability requirement is similar for comparative modes, and when the spacecraft cost is not too low as compared to the launch cost.

III. MAINTENANCE REQUIREMENTS ANALYSIS

In order to perform a complete analysis on the feasibility of a cost-effective maintenance system for automated spacecraft during the shuttle era, it was first necessary to establish a data basis to be used throughout the performance of the study. The performance of maintenance involves the correct combination of three separate and distinct elements: (1) the spacecraft upon which maintenance may be performed, (2) the system used to perform maintenance, and (3) the system used to transport both the spacecraft and the maintenance system.

This chapter discusses in some detail the process used to select the spacecraft upon which the performance of maintenance was investigated and describes a few of the important, maintenance related characteristics of a representative set of the selected spacecraft. Also discussed are the existing and currently planned capabilities of the separate STS elements, as they apply to maintenance. A brief description of typical maintenance mission scenarios follows, plus a brief description of functional and hardware requirements of maintenance resulting from combining the spacecraft, the maintenance systems, and the STS elements in the various scenarios.

A. MAINTENANCE APPLICABLE SET OF SPACECRAFT

Several hundred spacecraft missions will be flown during the shuttle era. In order to fully evaluate maintenance concepts, the spacecraft upon which maintenance may be performed must be considered. Rather than use all of the spacecraft in orbit as a basis for evaluating maintenance concepts, only those spacecraft upon which the performance of maintenance appears to be most feasible will be used in this study. This group of spacecraft is called the maintenance applicable set. Since that still could represent a large number of spacecraft, a smaller set of spacecraft, called the characteristic set, was chosen to represent the larger set. This characteristic set, which was selected to span most of the important maintenance-related characteristics of the larger, maintenance applicable set, was used where it was necessary to develop a greater level of detail than was possible for the entire maintenance applicable set.

Figure III-1 presents a brief, flow chart type summary of the approach used to select these two sets. The first step in the process was to determine the total automated spacecraft planned to be flown during the shuttle era. From this group of spacecraft, the spacecraft upon which it may prove feasible to perform maintenance and the spacecraft in the characteristic set would be selected.

The next step in the process involved the selection of the spacecraft upon which maintenance might prove to be more feasible by the elimination of all spacecraft upon which the performance of maintenance appeared to be very unlikely at this time. Several criteria were evaluated to be used in the elimination of spacecraft from maintenance considerations. The five criteria shown in Figure III-1 were the ones selected to be used in the

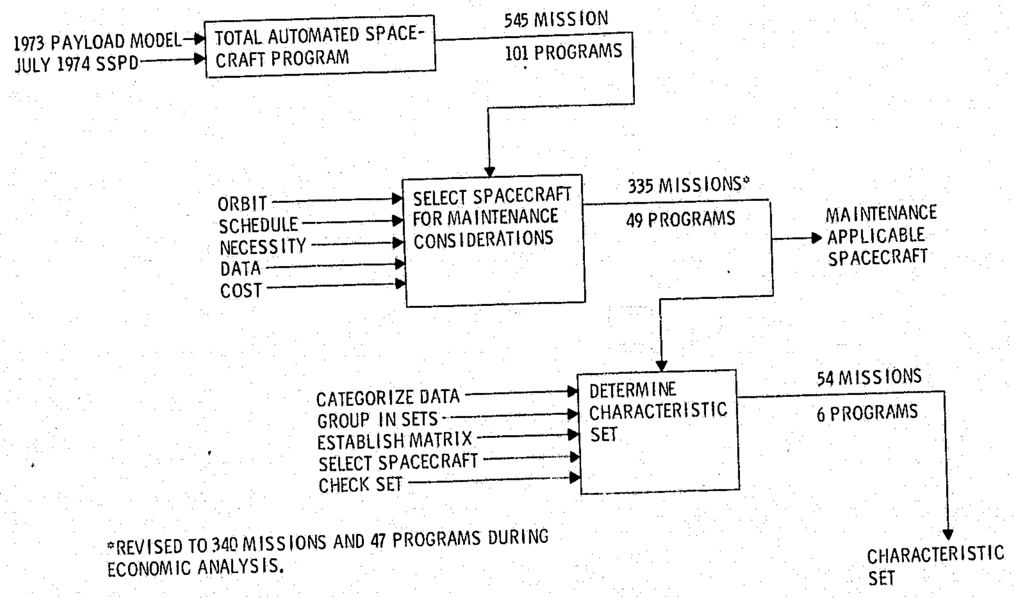


Figure III-1 Approach for Selection of Maintenance Applicable Spacecraft

elimination of spacecraft and the spacecraft remaining formed what is called the maintenance applicable set. The maintenance applicable set consists of all the spacecraft upon which it may prove feasible to perform maintenance and establishes the basis used to investigate the technical and economic interfaces between maintenance concepts and spacecraft. The characteristic set was selected from the maintenance applicable set and was used to represent the entire maintenance applicable set in several instances in the study.

The final step in the selection process involved the actual determination of the characteristic set. The criteria used to select the spacecraft were determined and a logical process was established to select the spacecraft. An iterative procedure was used to help verify that the set selected included the best mix of spacecraft to represent the maintenance applicable set.

The numbers shown as a part of the output from each step in Figure III-1 show the total number of spacecraft missions and programs for each step. There were 545 missions and 101 programs identified in the total automated spacecraft program. From these, 210 missions and 52 programs were eliminated to leave 335 missions and 49 programs in the maintenance applicable set. Finally, six spacecraft programs with a total of 54 missions were selected as the characteristic set of spacecraft.

In the economic analysis of the maintenance applicable set, a review of the data identified several inconsistencies. As a result, the spacecraft in the maintenance applicable set were revised to 47 spacecraft programs and 340 missions, which formed the actual basis for the cost analysis.

1. Selection of Maintenance Applicable Set

The total automated spacecraft were selected from two sources, the October 1973 NASA Payload Model (Chapter XI, Item A-1) and the July 1974 SSPD (Chapter XI, Item E-3). The payload model presented a summary of the future planned spaceflights for NASA for the years 1973 through 1991. While there are actually two SSPD documents, one for automated payloads and one for sortie lab (spacelab) payloads, only the automated payload SSPD was used. Similarly, only the automated payloads from the payload model were used. This study did not take into consideration any spacelab payloads. While it certainly could prove feasible to perform maintenance upon automated spacecraft during spacelab missions, maintenance upon spacelab payloads was not considered in this study. Also, no DoD payloads were considered in this study.

Table III-1 presents a summary of all of the automated spacecraft missions which formed a basis for this study. Spacecraft missions are listed by year of launch and by program. The table is basically the same

Table III-1 Total Automated Spacecraft Summary

YEAR OF LAUNCH -	73	74	75	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91	TOTAL
NASA																				
ASTRONOMY	2	2	2	1	2	4	2	6	6	9	11	9	15	14	12	12	12	9	13	143
PHYSICS	2	1	2	1	2	1	2	2	3	1	2	3	1	2	3	4	3	4	4	43
PLANETARY	1	1	2	2	2	2	5	2	7	0	3	4	5	5	2	0	2	2	2	49
LUNAR	0	0	0	0	0	0	1	0	0	0	0	1	0	1	1	1	1	1	1	8
LIFE SCIENCES	0	0	0	0	1	0	1	2	2	2	2	2	2	2	2	2	2	2	2	26
EARTH OBSERVATIONS	1	2	0	2	3	3	3	3	4	4	3	3	3	3	7	2	5	2	5	58
EARTH & OCEAN PHYSICS	0	1	0	1	1	0	2	2	4	2	0	0	1	4	0	0	0	4	0	22
COMMUN. & NAVIGATION	0	1	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	2
SPACE PROCESSING	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
SPACE TECHNOLOGY	0	0	0	0	0	0	0	1	0	1	0	1	0	1	0	1	0	1	0	6
TOTAL	6	8	7	7	11	10	16	18	26	19	21	23	27	32	27	22	25	25	27	357
NON-NASA/NON-DOD																				
COMMUN. & NAVIGATION	5	9	8	6	6	9	4	6	6	5	8	6	6	6	3	9	5	9	4	120
EARTH OBSERVATIONS	1	1	2	2	3	4	3	2	4	4	2	2	3	3	3	7	4	5	4	59
EARTH & OCEAN PHYSICS	0	0	0	0	0	0	0	0	0	0	0	0	0	3	0	3	0	3	0	9
TOTAL	6	10	10	8	9	13	7	8	10	9	10	8	9	12	6	19	9	17	8	188
GRAND TOTAL	12	18	17	15	20	23	23	26	36	28	31	31	36	44	33	41	34	42	35	545

SOURCES: THE OCTOBER 1973 PAYLOAD MODEL
THE JULY 1974 SSPD

as the summary table presented in the payload model, with the main exception being the additional missions added by the July 1974 SSPD. A total of 474 missions were shown in the payload model as compared to the 545 missions shown in this table.

Although this study encompasses the performance of maintenance upon shuttle era spacecraft, many of the missions shown in Table III-1 will be launched prior to the time that the shuttle and tug will be ready. In fact, some of the spacecraft shown have already been placed in orbit. These spacecraft were included in this study for two reasons. First, the format of this table was left essentially the same as the format of the summary table in the October 1973 payload model to enable an easier comparison between the two tables. This was done since the data from the summary table in the payload model has already been used in several related studies and it is necessary, when discussing results or making comparisons of studies, to know what the basis was for the studies. The second, and main reason why the pre-shuttle era spacecraft have been included was that it may prove feasible, and desirable, to perform maintenance upon spacecraft placed into orbit by expendable launch vehicles, after the shuttle and tug become operational.

At this point, 545 spacecraft missions, representing 101 spacecraft programs, had been identified. The next step in the process was to eliminate spacecraft upon which it was evident that it would probably not be feasible to perform maintenance. This would serve to narrow the boundary of the study and to help focus the major attention of this study upon the spacecraft most likely to benefit from orbital maintenance. Many criteria were considered to be used for the elimination of spacecraft. The following five were selected as valid criteria to be used:

1) Orbit - All spacecraft in earth escape orbits were eliminated from further consideration in this study. This included all lunar, planetary, and heliocentric spacecraft. While some maintenance could be performed on an earth escape mission while the spacecraft is still undergoing checkout in earth orbit, that is considered to be a part of the normal checkout capability of the shuttle or tug. Once the spacecraft has been placed on the escape trajectory, the energy or the time required to perform a maintenance mission would be excessive over the current planned STS capabilities. Once a maintenance concept has been selected, its use with an earth escape type of spacecraft could be considered; however, it would not be proper to base the selection of a maintenance concept, even in part, on requirements of a spacecraft designed for earth escape missions.

2) Schedule - All spacecraft planned on being launched prior to 1979 have been eliminated from this study. All spacecraft which would require a tug to emplace in orbit and which are planned on being launched prior to 1982 have been eliminated from this study. Initial operational capability (IOC) of the shuttle orbiter is expected to occur in 1980 and IOC of the full capability tug (ground-ruled for use in this study) is expected to occur in 1983. Even though the shuttle and tug will not be available to perform maintenance until these times, some payloads launched prior to these times have been included to permit consideration of performing maintenance on spacecraft already in orbit when the shuttle and full-up tug become operational. It was decided not to look at spacecraft launched more than 2 years before the IOCs of the shuttle and full-up tug since there probably will not be enough time for the maintenance concept design to affect the spacecraft, due to the long lead time usually required to implement design changes on a spacecraft. This will also help prevent levying too stringent requirements on the maintenance concept due to early spacecraft requirements.

3) Necessity - All spacecraft with no active systems (i.e., no electrical power system, no attitude control system, no tracking, telemetry and command, etc.) were eliminated from further consideration in this study. These spacecraft would be entirely passive and have no systems to maintain. Two spacecraft programs, with 8 missions (long duration exposure facility and MINILAGEOS) were eliminated under this criteria.

4) Inadequate Data - All spacecraft with inadequate data currently available were eliminated from further consideration in this study. Only one spacecraft was found that could be eliminated under this criteria, the space processing free flyer. Almost no other data, except the name of the spacecraft, were available, not even the number of missions.

5) Cost - All spacecraft for which a very simplified cost analysis showed that the expendable form of the spacecraft appeared to be much less expensive than the on-orbit maintenance form were eliminated from this study. Only the expendable and on-orbit maintainable spacecraft costs were compared for this simplified costing. For the simplified analyses, the following assumptions were used:

- On-orbit fleet size was one
- All shuttle and all tug flights had the same operations costs
- No loss factors
- The same cost sharing used for expendable as for on-orbit maintenance

An equation was set up comparing the total costs of the expendable form of the spacecraft program and the on-orbit maintainable program, and the number of missions required to make on-orbit maintenance less expensive than expendable was determined. Very simplified assumptions required that a large margin be included so as not to eliminate any spacecraft programs that should be considered.

Table III-2 presents a summary of all spacecraft eliminated at this time for all the five criteria. The data is presented by number of missions and number of programs eliminated for each criteria and totaled both by program and by criteria. If some missions were eliminated due to several criteria, only the first criteria under which it was eliminated is shown. If a certain spacecraft program was eliminated for more than

Table III-2 Summary of Spacecraft Eliminated

NASA PAYLOAD PROGRAM	ORBIT		LAUNCH <1979		TUG LAUNCH < 1982		NECESSITY		DATA		COST		TOTAL	
	MIS-SIONS	PRO-GRAM	MIS-SIONS	PRO-GRAM	MIS-SIONS	PRO-GRAM	MIS-SIONS	PRO-GRAM	MIS-SIONS	PRO-GRAM	MIS-SIONS	PRO-GRAM	MIS-SIONS	PRO-GRAM
ASTRONOMY	6	1	13	1	1	0	--	--	--	--	--	--	20	2
PHYSICS	9	3	9	0	4	0	--	--	--	--	--	--	22	3
PLANETARY	49	28	--	--	--	--	--	--	--	--	--	--	49	28
LUNAR	8	5	--	--	--	--	--	--	--	--	--	--	8	5
LIFE SCIENCES	--	--	1	0	--	--	--	--	--	--	--	--	1	0
EARTH OBSERVA-TION	--	--	11	2	1	0	--	--	--	--	1	1	13	3
EARTH AND OCEAN PHYSICS	--	--	3	2	2	0	2	1	--	--	4	3	11	6
NON-NASA/NON-DOD	--	--	56	0	21	0	--	--	--	--	1	1	78	1
COMMUNICATIONS/ NAVIGATION	--	--	2	2	--	--	--	--	--	--	--	--	2	2
SPACE PROCESSING	--	--	--	--	--	--	--	--	TBD	1	--	--	TBD	1
SPACE TECHNOLOGY	--	--	--	--	--	--	6	1	--	--	--	--	6	1
TOTAL	72	37	95	7	29	0	8	2	TBD	1	6	5	210	52

one criteria, the spacecraft program is listed under the criteria that eliminated the last mission, and thus the entire program. A total of 210 missions and 52 programs were eliminated. Table III-3 presents a summary of the spacecraft programs eliminated. This table does show which missions were eliminated for more than one criteria. For example, there were four missions in the total automated spacecraft (Table III-1) for the advanced synchronous meteorological satellite. Three of these four were to fly prior to 1979, so they were eliminated. Since they are to be in geosynchronous orbit, a Tug will be required. Since they fly before 1979, they could also be eliminated under the criteria "Tug Launch<1982". In Table III-2, these three missions were eliminated under the criteria "Launch< 1979". In Table III-3, these three missions are listed under both criteria. The one mission remaining was eliminated under the cost criteria. In Table III-2, this mission and the spacecraft type are both listed under the cost criteria.

Table III-3 Spacecraft Programs Eliminated

SPACECRAFT NAME	NUMBER OF MISSIONS ELIMINATED						
	NET	ORBIT	LAUNCH <1979	TUG LAUNCH <1982	NECES-SITY	DATA	COST
EXTRACORONAL LYMAN ALPHA EXPLORER	6	6		1			
ORBITING SOLAR OBSERVATORY	1		1				
HIGH ALTITUDE EXPLORER	6	6		1			
GRAVITY AND RELATIVITY SAT. - SOLAR	2	2					
HELIOCENTRIC & INTERSTELLAR SPACECRAFT	1	1					
EARTH RESOURCES TECHNOLOGY SATELLITE	1		1				
NIMBUS	2		2				
ADVANCED SYNCHRONOUS METEOROLOGICAL SAT.	4		3	3			1
GEODETIC EARTH ORBITING SATELLITE	1		1				
LASER GEODYNAMIC SATELLITE	1		1	1			
SEASAT	2		1				1
GEOPAUSE	2			1			1
MINILAGEOS	2				2		
MAGNETIC FIELD MONITOR SATELLITE	3			1			2
TIROS OPERATIONAL SATELLITE	7		6	1			1
APPLIED TECHNOLOGY SATELLITE	1		1	1			
COOPERATIVE APPLICATIONS SATELLITE	1		1	1			
LONG DURATION EXPOSURE FACILITY	6				6		
SPACE PROCESSING FREE-FLYER	TBD					TBD	
ALL PLANETARY MISSIONS	49	49	10	24			
ALL LUNAR MISSIONS	8	8		2			

The total number of spacecraft programs eliminated was 52, including 28 planetary programs, 5 lunar programs, and the 19 other programs listed in Table III-3. The 52 programs eliminated included 106 missions. The additional 104 missions eliminated came from spacecraft programs not entirely eliminated, but which have had some of their missions eliminated.

Table III-4 presents a summary of the spacecraft programs remaining once the criteria of orbit, schedule, necessity, inadequate data, and cost have been applied. At this point, a total of 335 missions and 49 spacecraft programs were left. This group of spacecraft was called the maintenance applicable set and represents all of the spacecraft which were considered for maintenance in the rest of this study.

Spacecraft programs and missions that have been eliminated from further consideration in this study were reevaluated later in the study to insure that none were eliminated that would actually be feasible for maintenance. This was performed once the detailed technical and economic analyses were well under way. Other criteria which were considered, but not selected, to be used to eliminate spacecraft at this time (such as size, weight,

Table III-4 Summary of Maintenance Applicable Set

PAYLOAD NO.	PAYLOAD MODEL CODE NO.	SPACECRAFT NAME
AS-03-A	AST-1B	COSMIC BACKGROUND EXPLORER
AS-05-A	AST-1C	ADVANCED RADIO ASTRONOMY EXPLORER
SO-03-A	AST-3	SOLAR MAXIMUM MISSION
HE-09-A	AST-4	LARGE HIGH ENERGY OBSERVATORY B
HE-03-A	AST-5A	EXTENDED X-RAY SURVEY
HE-08-A	AST-5B	LARGE HIGH ENERGY OBSERVATORY A
HE-10-A	AST-5C	LARGE HIGH ENERGY OBSERVATORY C
HE-05-A	AST-5D	HIGH LATITUDE COSMIC RAY SURVEY
AS-01-A	AST-6	LARGE SPACE TELESCOPE
SO-02-A	AST-7	LARGE SOLAR OBSERVATORY
AS-16-A	AST-8	LARGE RADIO OBSERVATORY ARRAY
HE-11-A	AST-9A	LARGE HIGH ENERGY OBSERVATORY D
HE-01-A	AST-9B	LARGE X-RAY TELESCOPE FACILITY
AS-07-A	AST-N1	3M AMBIENT TEMPERATURE IR TELESCOPE
AS-11-A	AST-N2	1.5M IR TELESCOPE
AS-13-A	AST-N3	UV SURVEY TELESCOPE
AS-14-A	AST-N4	1M UV - OPTICAL TELESCOPE
AS-17-A	AST-N5	30M IR INTERFEROMETER
HE-07-A	PHY-1A	SMALL HIGH ENERGY SATELLITE
AP-01-A	PHY-1B	UPPER ATMOSPHERE EXPLORER
AP-02-A	PHY-1C	EXPLORER-MEDIUM ALTITUDE
AP-04-A	PHY-2A	GRAVITATIONAL AND RELATIVITY SATELLITE - LEO
AP-05-A	PHY-3A	ENVIRONMENTAL PERTURBATION SATELLITE-A
AP-07-A	PHY-3B	ENVIRONMENTAL PERTURBATION SATELLITE-B
HE-12-A	PHY-5	COSMIC RAY LABORATORY
LS-02-A	LS-1	BIOMEDICAL EXPERIMENT SCIENTIFIC SATELLITE
EO-08-A	EO-3	EARTH OBSERVATORY SATELLITE
EO-09-A	EO-4	SYNCHRONOUS EARTH OBSERVATORY SATELLITE
EO-10-A	EO-5	APPLICATIONS EXPLORER (SPECIAL PURPOSE SATELLITE)
EO-12-A	EO-6	TIROS
OP-02-A	EOP-5	GRAVITY GRADIOMETER
OP-04-A	EOP-7	GRAVSAT
OP-05-A	EOP-8	VECTOR MAGNETOMETER SATELLITE
CN-51-A	NND-1	INTELSAT
CN-52-A	NND-2A*	DOMSAT A
CN-53-A	NND-2B	DOMSAT B
CN-58-A	NND-2C	DOMSAT C
CN-54-A	NND-3	DISASTER WARNING SATELLITE
CN-55-A	NND-4	TRAFFIC MANAGEMENT SATELLITE
CN-56-A	NND-5A	FOREIGN COMMUNICATION SATELLITE-A
CN-60-A	NND-5B*	FOREIGN COMMUNICATION SATELLITE-B
CN-59-A	NND-6	COMMUNICATIONS R&D PROTOTYPE
EO-56-A	NND-8	ENVIRONMENTAL MONITORING SATELLITE
EO-57-A	NND-9	FOREIGN SYNCHRONOUS METEOROLOGICAL SATELLITE
EO-58-A	NND-10	GEOSYNCHRONOUS OPERATIONAL METEOROLOGICAL SATELLITE
EO-61-A	NND-11	EARTH RESOURCES SURVEY OPERATIONAL SATELLITE
EO-59-A	NND-12	GEOSYNCHRONOUS EARTH RESOURCES SATELLITE
EO-62-A	NND-13	FOREIGN SYNCHRONOUS EARTH OBSERVATION SATELLITE
OP-51-A	NND-14	GLOBAL EARTH AND OCEAN MONITORING SYSTEM

*DROPPED FROM MAINTENANCE APPLICABLE SET IN ECONOMIC ANALYSIS.

types of subsystems, number of launches, on-board experiments, redundancy, etc.) were considered for use later in the study to help evaluate spacecraft/maintenance concept interfaces.

2. Selection of Characteristic Set

The next step in the selection process involved the establishment of the criteria that the characteristic set would satisfy and the establishment of the logical process used to select the characteristic set from the maintenance applicable set. The following criteria were established as conditions that the characteristic set should fulfill.

1) Taken from the maintenance applicable set.

2) Used to represent the entire maintenance applicable set in later tasks in this study.

3) Span all spacecraft maintenance-related data in the maintenance applicable set.

4) Show potential feasibility problems.

5) Cover all factors that could affect costs.

6) Permit evaluations of cost factors for spacecraft not in the characteristic set.

7) Have sufficient level of detail available.

8) Have high interest level in NASA, other government agencies, the aerospace industry, and in commerical satellite designer and user industries.

9) Span ranges of STS impacts.

In order to insure that all of the above criteria were satisfied as well as possible by the characteristic set, a logical process was established to select the spacecraft in the characteristic set. Figure III-2 presents a flow chart of that process.

Step 1 - Categorize the important maintenance-related characteristics of spacecraft in the characteristic set. In order to ensure that the characteristic set did span all maintenance-related data in the maintenance applicable set, that all factors that could affect cost were covered, and that the costing evaluation could be spread across the entire maintenance applicable set, it was necessary to first, determine what type of data would be required on spacecraft in the maintenance applicable set, second, to gather the data for all spacecraft in the maintenance applicable set,

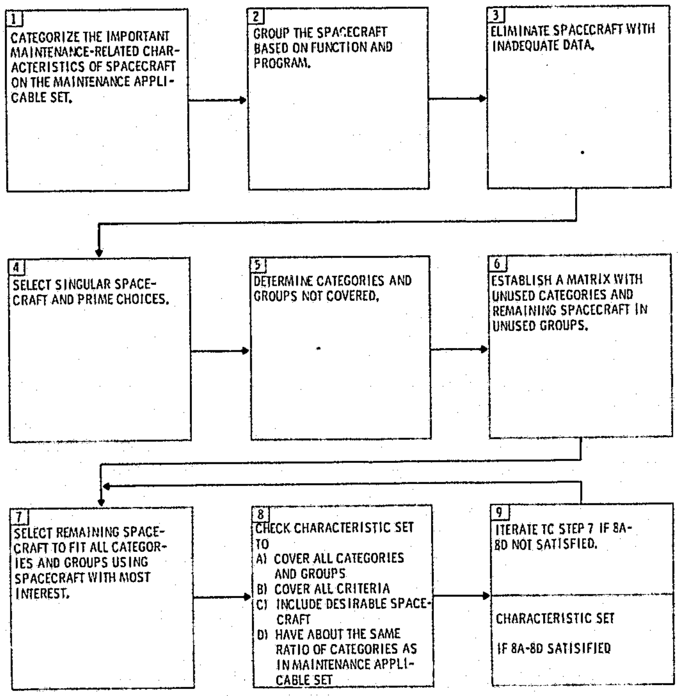


Figure III-2 Selection Process for Characteristic Set

and then to categorize the data for each spacecraft. The categorization of data would enable a more orderly investigation of the effects of that data. Many different categories of data were investigated to determine what the effects might be upon maintenance concepts. Some categories were discarded as having no real effect on maintenance concepts, or only minor effects. The categories finally selected were as follows:

1) Orbit - Could maintenance be performed from the shuttle, or is a tug also required?

2) Size - Is the spacecraft small, medium, or large? Size is not only a measure, somewhat, of spacecraft and system complexity, but size also affects STS transportation capabilities, and thus gives some measure of STS impacts.

3) Attitude - What is meant here by attitude dynamics is the spacecraft stable or spinning? While this study did not attempt to solve the problems associated with performing maintenance upon a spinning satellite, this could present a potential feasibility problem and should be investigated.

4) Number of Missions - As the simplified cost analysis showed, the number of missions (or number of operating cycles) would be one of the most important economic parameters to consider.

5) On-Orbit Fleet Size - As was also recognized in the simple cost analysis discussed above, on-orbit fleet size was an important parameter that could certainly affect the economics of a maintenance program.

6) Life - Satellite lifetime would also be an important factor in the feasibility study of on-orbit maintenance. Different requirements might be necessary for a satellite with a few months lifetime as compared to a satellite with a 10 year lifetime.

7) Cost - Satellite costs, both for DDT&E and for each satellite, are two of the prime cost parameters used to determine the economic feasibility of any satellite program.

8) First Launch Date - The first launch date of a spacecraft is important for two reasons. First of all, the time that the spacecraft will first be launched, as compared to the different IOCs of the STS elements, will be important. And, secondly, the time when the spacecraft must first be launched is important as far as user (satellite designers, builders, and customers) acceptance of maintenance concepts.

Data on these categories were gathered for all 49 spacecraft programs in the maintenance applicable set. The data were then evaluated to determine the most advantageous divisions in the various categories to best represent the maintenance applicable set. The divisions decided upon are described as follows:

1) Orbit - The best division of this category was to divide the spacecraft into those delivered into orbit by the orbiter and those which also require a tug. All spacecraft in the maintenance applicable set used orbits such that if a tug was required to place the satellite in orbit, a tug would be required to perform maintenance.

2) Size - Weight of the spacecraft was used as the best approximation of size. Most of the spacecraft were about the same density; those that weren't were well within one order of magnitude. A plot was made arranging all of the spacecraft according to weight (see Figure III-3). This figure shows all the spacecraft arranged from the heaviest to the lightest. The spacecraft were then divided into groups to try to place approximately an

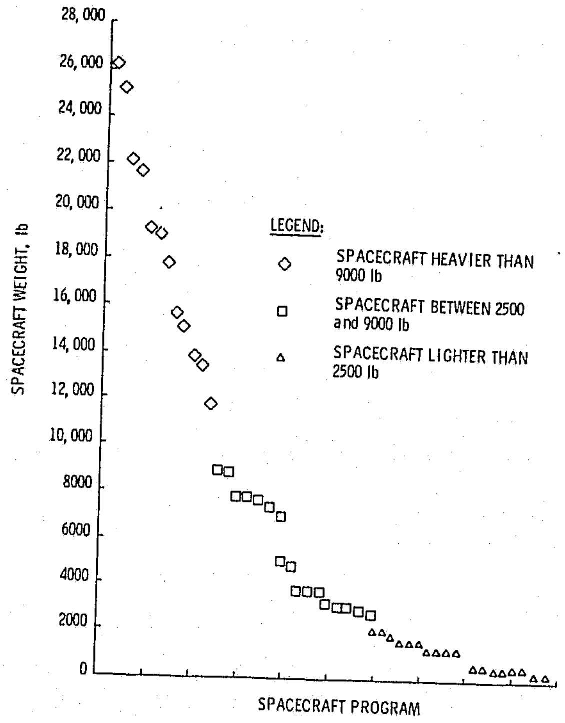


Figure III-3 Spacecraft Weight Plot

equal number of spacecraft in each division, but at the same time to try to use natural divisions between groups. This technique led into dividing this category into spacecraft heavier than 9,000 pounds, spacecraft between 2,500 and 9,000 pounds, and spacecraft lighter than 2,500 pounds. The 2,500 pound value was also selected since it is the approximate tug round trip capability to geosynchronous orbit.

3) Attitude - The only division made here was into spinning and 3 axis stabilized spacecraft.

4) Number of Missions - A plot was also made arranging the spacecraft according to number of missions (see Figure III-4). Since single missions had already been shown to be an important division in the simplified cost analysis, this was selected as one division. The rest of the spacecraft types were then divided so that about half fell into each division. This led to the divisions of single missions, two to five missions, and six or more missions.

5) On-Orbit Fleet Size - More than half of the spacecraft types had an on-orbit fleet size of one. Of those remaining, most were either two or three, with only a few being four or more. These were the divisions selected.

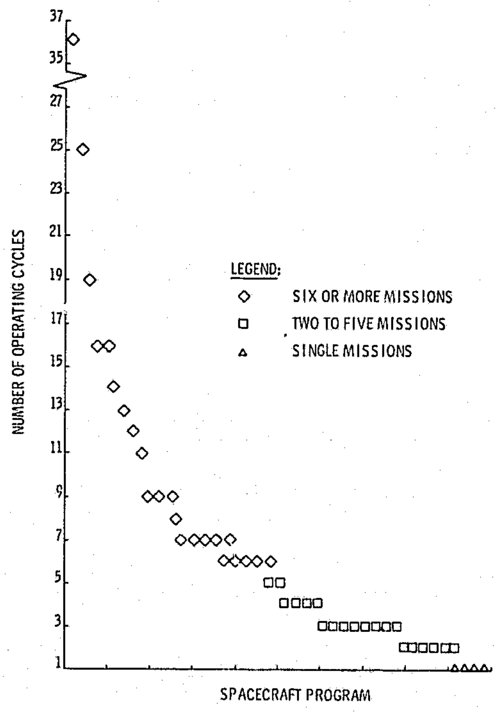


Figure III-4 Spacecraft Operating Cycle Plot

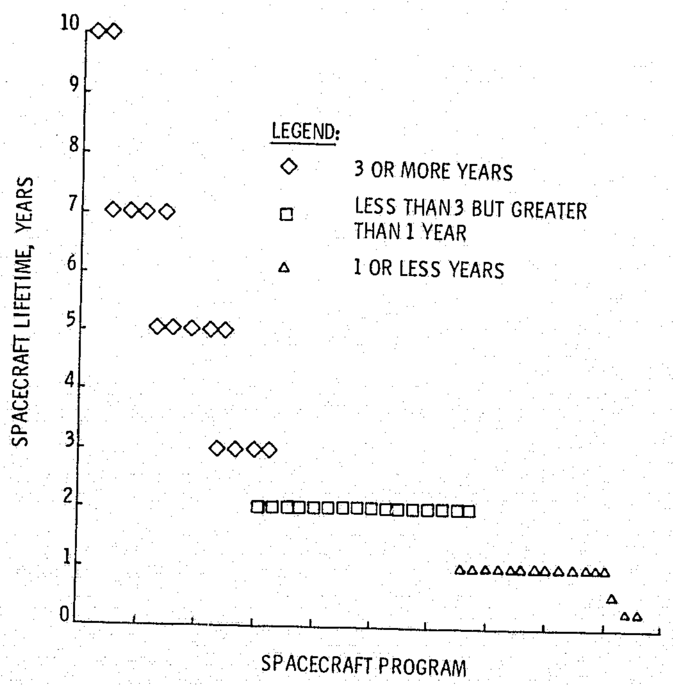


Figure III-5 Spacecraft Lifetime Plot

6) <u>Spacecraft Lifetime</u> - A plot was made arranging the spacecraft according to spacecraft lifetime (see Figure III-5). The spacecraft were divided into three equal divisions with lifetimes of one year or less, one to three years, and three years or more.

7) <u>Non-Recurring Cost</u> - This category lent itself into an easy division with half the spacecraft programs being less than 100 million 1972 dollars (M) and half being 100M or more.

8) <u>Recurring Cost</u> - Division of this category along natural break points accounted for divisions of spacecraft less than \$15M, spacecraft between \$15 and \$50M, and spacecraft greater than \$50M.

9) <u>First Launch Date</u> - Natural division points here led to two divisions; spacecraft launched from 1979 through 1983 and spacecraft launched in 1984 or later. This is also the point at which the full capability tug will be available.

Table III-5 presents a summary of the categorization results obtained with both spacecraft programs and number of total missions included in each category division. Table III-6 presents a summary of all the spacecraft in the maintenance applicable set, showing the categorization of data.

Table III-5 Categorization Summary

CATEGORY	DIVISION	NUMBER OF SPACECRAFT PROGRAMS	NUMBER OF SPACECRAFT MISSIONS
ORBIT	ORBITER ORBITER/TUG	27 22	222 113
WEIGHT	< 2500 lb 2500 < SPACECRAFT < 9000 lb < 9000 lb	19 17 12	132 107 87
ATTITUDE DYNAMICS	SPINNING 3-AXIS	6 41	21 280
NUMBER OF MISSIONS	1 2-5 ≥ 6	4 22 23	4 69 262
MAXIMUM ON-ORBIT FLEET SIZE	1 2-3 ≥ 4	27 16 6	169 110 56
SPACECRAFT LIFETIME	≤ 1 YEAR 1 YEAR < LIFETIME < 3 YEARS ≥ 3 YEARS	16 17 15	154 92 80
NONRECURRING COST	≥ \$100M < \$100M	21 21	102 153
RECURRING COST	< \$15M \$15M ≤ COST ≤ \$50M > \$50M	16 16 10	111 84 60
FIRST LAUNCH DATE	1979 THROUGH 1983 ≥ 1984	32 17	260 75

Table III-6 Categorization of Data for Maintenance Applicable Set

PAYLOAD CODE NO.		ORBIT		WEIGHT, lb			ATTI-TUDE		NUMBER OF MISSIONS			ON-ORBIT FLEET SIZE			LIFE-TIME, yr			NONRE-CURRING COST		RECURRING COST			FIRST LAUNCH DATE	
SSPD	P.M.	ORBITER	ORBITER/TUG	<2500	2500-9000	>9000	SPIN	3-AXIS	1	2-5	≥6	1	2-3	≥4	≤1	1-3	≥3	<$100M	>$100M	<$15M	$15M - $50M	>$50M	79-83	≥84
AS-03	AST-1B	X		1312				X			7	1			1			X		X			79	
AS-05	AST-1C		X		2644			X		3			2				3	X		X			83	
SO-03	AST-3	X		1678				X			6	1				2			X		X		80	
HE-09	AST-4	X				13792		X		2		1				2							80	
HE-03	AST-5A	X				17664		X		3		1				2			X			X	82	
HE-08	AST-5B	X				19183		X		3		1				2			X			X		86
HE-10	AST-5C	X				11766		X		2		1				2			X			X		87
HE-05	AST-5D	X				15572		X	1			1					3		X			X		91
AS-01	AST-6	X				25005		X			12	1			1								80	
SO-02	AST-7	X				21664		X			7	1				2			X			X		85
AS-16	AST-8		X		2867			X		4		1				2		X			X			85
HE-11	AST-9A	X				14930		X		4		1				2			X			X	83	
HE-01	AST-9B	X				26171		X		3		1				2			X			X		86
AS-07	NEW-1	X				18950		X			9	1			1								83	
AS-11	NEW-2	X				13329		X			36	1			.25								81	
AS-13	NEW-3	X			7563			X			6	1			.25								80	
AS-14	NEW-4	X			8922			X			11	1			.25								81	
AS-17	NEW-5	X			7695			X		4		1			1									85
HE-07	PHY-1A	X		1311				X			6	1			1			X		X			81	
AP-01	PHY-1B		X	2004			X			2		1			1			X		X				85
AP-02	PHY-1C		X	599			X			3		1			1				X	X			83	
AP-04	PHY-2A	X		1323				X		2		1			1			X			X		80	
AP-05	PHY-3A		X		3281			X	1			1					3		X		X			84
AP-07	PHY-3B		X		8701			X		2		1					3		X		X			87
HE-12	PHY-5	X				22121		X		5		1			1				X			X		87
LS-02	LS-1	X		1504							25	1			.5			X		X			79	
EO-08	EO-3	X			7662			X			19		2			2			X			X	79	
EO-09	EO-4		X		3376			X			8		2			2			X		X		83	
EO-10	EO-5	X		310				X			16		2		1			X		X			79	
EO-12	EO-6		X		4741			X		2		1				2		X			X		82	
OP-02	EOP-5	X			7226		X		1			1			1				X		X		80	
OP-04	EOP-7	X			6805			X	1				2			2			X		X		79	
OP-05	EOP-8	X		310				X			9		3		1			X		X			81	
CN-51	NN/D-1		X		3246			X			16			16			10	X			X		83	
CN-52	NN/D-2A		X	577			X			3			3				7	X		X			82	
CN-53	NN/D-2B		X		3246			X			14			14			10	X			X			84
CN-58	NN/D-2C		X	1913				X			6			6			7	X		X			83	
CN-54	NN/D-3		X	1285				X		3			3				5	X		X			82	
CN-55	NN/D-4		X	658				X			6			6			5	X		X			82	
CN-56	NN/D-5A		X	679				X		3			3				7	X		X			82	
CN-60	NN/D-5B		X	732				X			7			7			7	X		X				85
CN-59	NN/D-6		X	2109				X		3			3				5		X		X			85
EO-56	NN/D-8		X		4860			X			7		2			2		X			X		82	
EO-57	NN/D-9		X	566			X			5			3				5	X		X			82	
EO-58	NN/D-10		X	566			X				7			5			5	X		X			82	
EO-61	NN/D-11	⊗		1616				X			13		2			2			X			X	79	
EO-59	NN/D-12		X		3376			X		4			2			2			X		X			88
EO-62	NN/D-13		X		3376			X		4			3			2			X		X			88
OP-51	NN/D-14	X									9		3						X		X			86

⊗ LATER MOVED TO ORBITER/TUG COLUMN

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Step 2 - Group the spacecraft based on function and program - The spacecraft had already been partially grouped according to payload program in the payload model. There was a little overlap in function between the groups, and a slightly difference grouping was used in the SSPD. A very simplified grouping was used here, as follows:

- Medium/small space viewing observatories
- Large space viewing observatories
- Earth viewing, experimental
- Life sciences
- Commercial communications
- Commercial earth viewing

Six groups were formed, and since one spacecraft from each group was to be selected, this set the number of spacecraft types in the characteristic set at six. Table III-7 presents a summary of the groupings showing the spacecraft in each group.

Table III-7 Grouping of Spacecraft for Characteristic Set

MEDIUM/SMALL SPACE VIEWING OBSERVATORIES	LARGE SPACE VIEWING OBSERVATORIES	EARTH VIEWING EXPERIMENTAL	LIFE SCIENCES	COMMERCIAL COMMUNICATION	COMMERCIAL EARTH VIEWING
AST-1B	AST-4*	EO-3†	LS-1	NN/D-1	NN/D-8
AST-1C	AST-5A	EO-4		NN/D-2A	NN/D-9
AST-3	AST-5B	EO-5		NN/D-2B	NN/D-10
AST-8*	AST-5C*	EO-6		NN/D-2C	NN/D-11
NEW-3*	AST-5D*	EOP-5		NN/D-3	NN/D-12*
NEW-4*	AST-6*	EOP-7		NN/D-4	NN/D-13*
NEW-5*	AST-7*	EOP-8		NN/D-5A	NN/D-14*
PHY-1A	AST-9A			NN/D-5B*	
PHY-1B	AST-9B			NN/D-6*	
PHY-1C	NEW-1*				
PHY-2A	NEW-2*				
PHY-3A	PHY-5*				
PHY-3B					

*TOTAL DATA NOT AVAILABLE.
†GROUNDRULED OUT FOR CHARACTERISTIC SET.

Step 3 - Eliminate spacecraft with inadequate data - In order to ensure that the spacecraft in the characteristic set constituted the best span of data, and to ensure that each one had sufficient detail available, all spacecraft for which complete data were not available were eliminated from consideration for the characteristic set (they were still kept in the maintenance applicable set). This included any spacecraft for which there were blank columns in Table III-6, or any spacecraft which were not in the detailed descriptions (Level B) of the SSPD. Those spacecraft with inadequate data are so indicated in Table III-7 by an asterisk.

Step 4 - Select singular spacecraft and prime choices - The first spacecraft for the characteristic set were selected at this time. The biomedical experiments scientific satellite (BESS) was the only spacecraft in the life sciences group and was selected as a singular spacecraft. The international communications satellite (INTELSAT) was selected as a prime choice from the commercial communications group. This was done since: 1) INTELSAT had the highest number of missions in the commerical communications group, 2) INTELSAT appears almost identical to DOMSAT B and thus, between the two, 30 of the 61 spacecraft in this group could be represented by INTELSAT, 3) INTELSAT is an ongoing program and more data exists on it than any other spacecraft in the group, and 4) COMSAT, the operator for the INTELSAT system is performing a concurrent study to this one and could furnish data to this study besides having a high interest in the investigation of the feasibility of maintenance for INTELSAT.

Step 5 - Determine categories and groups not covered - These two spacecraft being in the characteristic set covered, of course, the groups of life sciences and commercial communications. Four more spacecraft were to be selected, one each from the large space viewing observatories, the medium/small space viewing observatories, the earth viewing experimental and the commercial earth viewing groups. The divisions in the categories not covered include: weight > 9,000 lbs; attitude, spinning; number of missions, 1, 2-5; on-orbit fleet sizes, 2-3; life, 1-3; non-recurring cost > \$100M; recurring cost > \$50M; and first launch date, ≥ 84.

Step 6 - Establish a matrix with unused categories and remaining spacecraft in unused groups - A matrix was established (see Table III-8) to enable the rest of the spacecraft to be picked for the characteristic

Table III-8 Spacecraft Selection Matrix

CATEGORIES	GROUPS: MEDIUM/SMALL SPACE VIEWING OBSERVATORIES	LARGE SPACE VIEWING OBSERVATORIES	EARTH VIEWING EXPERIMENTAL	COMMERCIAL EARTH VIEWING
WEIGHT: >9000 lb	0	AST-5A AST-5B AST-9A AST-9B	0	0
ATTITUDE: SPINNING	PHY-1B PHY-1C	0	EOP-5	NN/D-9 NN/D-10
NUMBER OF MISSIONS: 1	PHY-3A	0	EOP-5 EOP-7	0
NUMBER OF MISSIONS: 2-5	AST-1C PHY-1B PHY-1C PHY-2A PHY-3B	AST-5A AST-5B AST-9A AST-9B	EO-6	NN/D-9
ON-ORBIT FLEET SIZE: 2-3	AST-1C	0	EO-4 EO-5 EOP-7 EOP-8	NN/D-8 NN/D-9 NN/D-11
LIFETIME: 1-3	AST-3	AST-5A AST-5B AST-9A AST-9B	EO-4 EO-6 EOP-7	NN/D-8 NN/D-11
NONRECURRING COST: > $100M	AST-3 PHY-1C PHY-3A PHY-3B	AST-5A AST-5B AST-9A AST-9B	EO-4 EOP-5 EOP-7	NN/D-11
RECURRING COST: > $50M	0	AST-5A AST-5B AST-9A AST-9B	0	NN/D-11
FIRST LAUNCH DATE: ≥ 84	PHY-1B PHY-3A PHY-3B	AST-5B AST-9B	0	0

set. This matrix included all the spacecraft left in each group that had not had a selection, listed for each category that was still open. Not included were any spacecraft from the life sciences or commercial groups, nor were any spacecraft included for which total data was not available. Also excluded was EO-3, the earth observatory satellite (EOS) which had been ground-ruled out of in the characteristic set.

Step 7 - Select remaining spacecraft

Step 8 - Check characteristic set

Step 9 - Iterate back to Step 7 until check in Step 8 is satisfied - These three (and final) steps were carried out together. Several iterations were carried out to select the best possible choices for the characteristic set. The final selection for the characteristic set included the following spacecraft:

Payload Number	Payload Model Code Number	Spacecraft
HE-01-A	AST-9B	Large X-Ray Telescope Facility (LXRT)
AP-01-A	PHY-1B	Upper Atmosphere Explorer (UAE)
LS-02-A	LS-1	Biomedical Experiments Scientific Satellite (BESS)
OP-04-A	EOP-7	Gravity Satellite (GRAVSAT)
CN-51-A	NN/D-1	International Communications Satellite (INTELSAT)
EO-56-A	NN/D-8	Environmental Monitoring Satellite (EMS)

Table III-9 presents a summary of the spacecraft chosen for the characteristic set. The table shows how the characteristic set spans the entire range of categories used in the process of determining the characteristic set. The two lines at the bottom of the table give the number of spacecraft programs in the characteristic set for each category and the number of spacecraft programs in the maintenance applicable set for each category. In most cases, the ratios are approximately the same.

3. Description of Spacecraft in Characteristic Set

Figure III-6 presents a schematic showing the configurations of the six spacecraft selected for the characteristic set. The wide variety of sizes, shapes, and appendages of the six spacecraft are evident. The

Table III-9 Characteristic Set Summary

			WEIGHT, lb			ATTITUDE		NUMBER OF MISSIONS			ON-ORBIT FLEET SIZE			LIFETIME (years)			NON-RECURRING COST		RECURRING COST			FIRST LAUNCH DATE	
SPACECRAFT	ORBITER	TUG	<2500	2500-9000	>9000	SPIN	3-AXIS	1	2-5	≥6	1	2-3	≥4	≤1	1-3	≥3	<$100M	≥$100M	<$15M	$15M - $50M	>$50M	1979-1983	≥1984
LARGE X-RAY TELESCOPE FACILITY (LXRT)	X				X		X		X		X				X			X			X		X
UPPER ATMOSPHERE EXPLORER (UAE)		X	X			X			X		X			X			X		X			X	
BIOMEDICAL EXPERIMENTAL SCIENTIFIC SATELLITE (BESS)	X		X			(GRAV GRAD)				X		X		X			X		X			X	
GRAVITY SATELLITE (GRAVSAT)	X			X			X	X				X			X			X		X		X	
INTERNATIONAL COMMUNICATION SATELLITE (INTELSAT)		X		X			X			X			X			X	X			X		X	
ENVIRONMENTAL MONITORING SATELLITE (EMS)		X		X			X			X		X			X		X			X		X	
TOTAL CHARACTERISTIC SET	3.	3	2	3	1	1	4	1	2	3	2	3	1	2	3	1	4	2	2	3	1	5	1
APPLICABLE SPACECRAFT	27	22	19	17	12	6	41	4	22	23	26	17	6	16	17	15	21	21	16	16	10	32	17

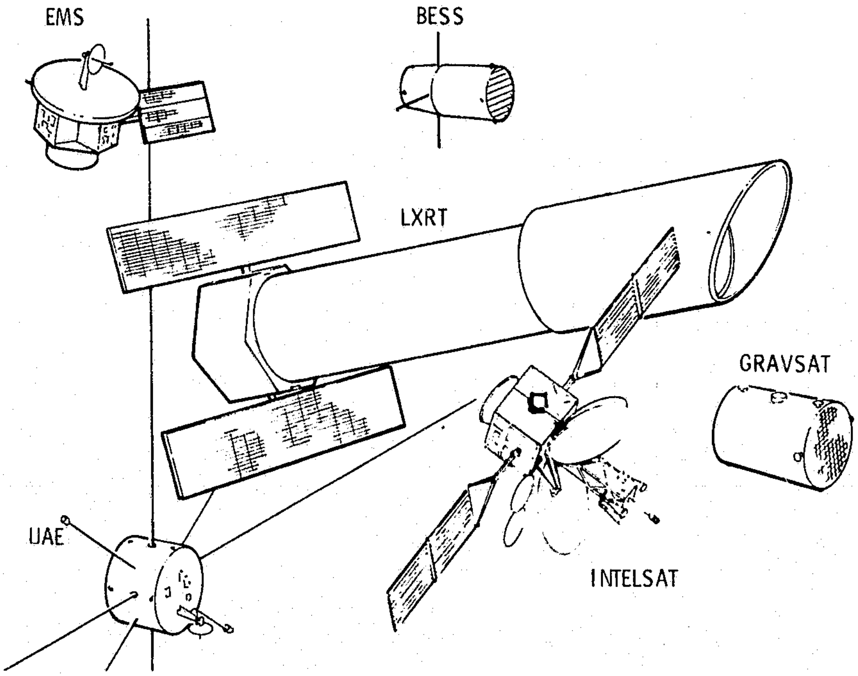

Figure III-6 Configurations of Characteristic Set

checks used in Step 8 of the selection process helped to ensure that the characteristic set did form the best representation of the spacecraft in the maintenance applicable set.

The characteristic set was used in several later steps in this study, mainly to investigate more detailed interfaces between spacecraft and maintenance systems. In particular, it was used to investigate various modularization schemes for spacecraft. During and following the economic evaluation, more attention was paid to all of the spacecraft in the maintenance applicable set to ensure that the total economic effects of maintenance were evaluated.

Table III-10 presents a summary of some important maintenance-related parameters for the six spacecraft in the characteristic set. The parameters presented were mainly useful in the technical considerations of maintenance, although many of these same parameters were also considered under the economic evaluation of maintenance.

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Table III-10 Spacecraft Requirements and Capabilities

Spacecraft Requirements and Capabilities	Spacecraft					
	LXRT	UAE	BESS	GRAVSAT	INTELSAT	EMS
Orbit	Incl. - 0, 15, 28.5 deg Apogee - 240, 250, 260 n mi Perigee - 240, 250, 260 n mi Delivery - orbiter Launch site - ETR	Incl. - 70, 90, 110 deg Apogee - 1800, 1900, 2000 n mi Perigee - 100, 140, 180 n mi Delivery - orbiter/tug Launch site - WTR	Incl. - TBD, 37.7, TBD deg Apogee-270, 300, TBD n mi Perigee-270, 300, TBD n mi Delivery-orbiter Launch site-any	Incl.-89.9, 90, 90.1 deg Apogee-159, 162, 165 n mi Perigee-159, 162, 165 n mi Delivery-orbiter Launch site-WTR	Incl.- -0.1, 0, +0.1 deg Apogee- 19,298 19,323, 19,348 n mi Perigee- 19,298 19,323, 19,348 n mi Delivery-orbiter/tug Launch site-ETR	Incl.-102.43, 102.97, 103.1 deg Apogee-890, 915, 940, n mi Perigee-880, 905, 940 n mi Delivery-orbiter/tug Launch site - WTR
Weight	Total spacecraft weight, launch = 21,877 lb; retrieve = 20,655 lb; C&D equipment in orbiter = 265 lb	Total spacecraft weight, launch = 2,004 lb; retrieve = 1,618 lb; C&D equipment in orbiter = 251 lb	Total spacecraft weight, launch = 5,000 lb; retreive = 4,000 lb	Total spacecraft weight, launch*= 6,804 lb; retrieve*= 6,196 lb; C&D equipment in orbiter = 100 lb *weight given for two s/c, both launched on same flight	Total spacecraft weight, launch = 2,090 lb; retrieve = 1,725 lb; C&D equipment in orbiter = 100 lb	Total spacecraft weight, launch = 4,860 lb; retrieve = 4,833 lb; C&D equipment in orbiter = 100 lb
Current Definition of Spacecraft	Ground refurbishable and on-orbit maintainable, capability to include docking (but not shown in weight)	Expendable, but capability exists to include docking	Ground refurbishable, capability to include dock-	Expendable, no current capability for docking	Expendable, no current capability for docking	On-orbit maintainable, capability to include docking
Attitude Dynamics	3-axis stable	Spinner	Gravity gradient	3-axis stable	3-axis stable	3-axis stable
Spacecraft Configuration	See Fig. III-6	See Fig. III-6	See Fig III-6	See Fig. III-6	See Fig. III-6	See Fig. III-6
Docking/Capture/Attachment Mechanisms	Capability exists to include a 7-ft diameter x 1.5-ft long docking adapter of TBD lb	Capability exists but none currently provided		None	None	Capability exists, but none currently provided
Consumables	1000 lb hydrazine, (propulsion), 4 tanks; 220 lb GN_2, at least two tanks (ACS)	370 lb hydrazine (propulsion), TBD tanks; 15 lb hydrazine (ACS); TBD tanks	Water Hydrazine Food	304 lb GN_2 and NH_3 system (ACS) TBD tanks each spacecraft	365 lb hydrazine, 4 tanks	27.5 lb hydrazine (propulsion) TBD tanks; TBD lb GN_2 (ACS) TBD tanks
Constraints on Maintenance Concepts	None	None	None	None	Waveguide connectors; radiator area alignment problems	None
Protective Covers	Contamination and thermal protective cover Contamination protective cover	Contamination protective cover		None	None	None
Safety Critical Items	Cryogenics Heat Rejection capability of SSM Pressurized shell around	Pyrotechnics NiCd batteries	High-pressure bottles	Momentum wheel Batteries	Pyrotechnics Pressure Tanks	Pressurized tanks Momentum wheels Batteries Pyrotechnics

[illegible]on-Mainten- ance Concepts					area alignment problems	
Protective Covers	Contamination and thermal protective cover Contamination protective cover	Contamination protective cover		None	None	None
Safety Critical Items	Cryogenics Heat Rejection capability of SSM Pressurized shell around payload Control moment gyros Batteries	Pyrotechnics NiCd batteries	High-pressure bottles	Momentum wheel Batteries	Pyrotechnics Pressure Tanks	Pressurized tanks Momentum wheels Batteries Pyrotechnics
Constraints on Orientation	Operating: tele- scope points to- ward space, solar arrays perpen- dicular to sun axis Nonoperating: solar arrays perpendicular to sun axis			Nonoperating: none Operating: X-axis along velocity vector body solar arrays, no re- quirements	Nonoperating: so- lar arrays per- pendicular to sun axis Operating: earth pointing, ±0.1 deg about each axis	Operating: earth pointing for ex- periments OEO- 011 and OEO-012, solar array per- pendicular to sun axis Nonoperating: solar array per- pendicular to sun axis
Acceleration	Operating: 1 x 10^{-4} g Nonoperating: 5g	Operating: 1.0g Nonoperating: 15g		Operating: 0.1g Nonoperating: 3.5g		Operating: 0.1g Nonoperating: 3.5g
Power Re- quirements and Capa- bilities	Requirements, operating: Mission equip- ment: Peak: 905 w St.St.: 665 w Subsystems: Peak: St.St.: Requirements, nonoperating: Mission equip- ment: Peak: St.St.: 229 w Subsystems: Peak: St.St.: Capability: Peak: 1800 w St.St.: 1200 w Batteries: 4.8 kW/hr	Requirements, operating: Mission equip- ment: Peak: 62 w St.St.: 62 w Subsystems: Peak: St.St.: Requirements, nonoperating: Mission equip- ment: Peak: St.St.: 0 w Subsystems: Peak: St.St.: Capability: Peak: St.St.: 200 w Batteries: 40 amp-hr		Requirements, operating: Mission equip- ment: Peak: St.St.: 75 w Subsystems: Peak: St.St.: Requirements, nonoperating: Mission equip- ment: Peak: St.St.: Subsystems: Peak: St.St.: Capability: Peak: St.St.: 325 w Batteries:	Requirements, operating: Mission equip- ment: Peak: St.St.: Subsystems: Peak: St.St.: Total Peak: St.St.: 1000 w Requirements, nonoperating: Total: Peak: St.St.: 150 w	Requirements, operating: Mission equip- ment: Peak: St.St.: 219 w Subsystems: Peak: St.St.: Requirements, nonoperating: Mission equip- ment: Peak: St.St.: Subsystems: Peak: St.St.: Capability: Peak: St.St.: 589 w Batteries: 20 amp-hr
Data Require- ments and Capabilities	Data require- ments, mission equipment: Science: 4.465×10^5 bps D 3 Hz A Housekeeping: 2.042×10^3 bps D Command: 1.024×10^3 bps D Subsystems: Science: Housekeeping: Command: Data Capability: Storage: 287 mbits Transmission: S-band, 5.12x $10^4 + 1 \times 10^6$ bps	Data require- ments, mission equipment: Science: 9.810×10^3 bps D 9.05×10^4 bps A Housekeeping: 0 Command: 0 Subsystems: Science: Housekeeping: Command: Data capability storage: 64K words Transmission: 10 watt S-band 5 watt UHF	S-Band	Data require- ments, mission equipment: Science: 5bps D Housekeeping: 1k bps D Command: 48bps D Subsystems: Science: Housekeeping: Command: Data Capability Storage: Transmission S-band 1×10^3 bps	Data require- ments, total 1000 bps	Data require- ments, mission equipment: Science: 5.597x10 bps D Housekeeping: 20+TBD bps D .5 Hz 38 channel A Command: 20+TBD bps D Subsystems: Science: Housekeeping: Command: Data capability Storage: Transmission: S-band 2×10^4 bps X-band 2×10^8 bps

B. STS MAINTENANCE CAPABILITIES

The current known capabilities of the STS, as related to maintenance, have been listed in this section. The various elements of the STS investigated in this section include the orbiter, the full-capability tug, and the ground support elements. The data was primarily obtained from the documents listed in Chapter XI as Items D-1 to D-3, E-5 to E-8, and J-16 and -17. Although the STS will probably eventually include such elements as the tracking and data relay satellite system (TDRSS), some form of an interim upper stage (IUS) and an earth orbital teleoperator system (EOTS), data on the current capabilities of these systems are not as readily available and have not been listed in this section. Considerations of the use of those elements are discussed in other chapters and sections of this report, and estimated capabilities are presented where required.

Table III-11 presents a brief description of the types of data considered for maintenance-related STS capabilities for the three STS elements investigated. Tables III-12, III-13, and III-14 present detailed

Table III-11 Maintenance-Related STS Capabilities Summary

	ORBITER	TUG	GROUND SUPPORT
PAYLOAD SIZE & HANDLING	X	X	X
GUIDANCE & NAVIGATION	X	X	
ATTITUDE CONTROL SYSTEM	X	X	
TRACKING, TELEMETRY, & COMMAND	X	X	X
ELECTRICAL POWER SYSTEM	X	X	X
THERMAL CONTROL SYSTEM	X	X	X
FLUIDS	X	X	X
EVA / IVA	X		
SHUTTLE MANIPULATOR SYSTEM	X		
OPERATIONAL	X	X	X

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Table III-12 Orbiter-Maintenance Related Capabilities

Category	Capabilities
Payload Size and Handling	• Payload bay size - 60 ft x 15 ft diameter, less 7.7 ft for docking module, less 9.7 ft for OMS kits. • Payload attachment fittings - twelve 3-point attachments and one 2-point attachment. All except the first three and last two are spaced 59 in. apart. • Active retention and release mechanism for the payload • Payload delivery - 65,000 lb to 100 n mi at 28.5 deg inclination. Other altitude, inclination, weight capabilities as shown in Chapter XI, Item D-2. • Payload return - 32,000 lb, maximum--other capabilities as shown in Chapter XI, Item D-2.
Guidance and Navigation	• Orbiter to payload - state vector, attitude, GMT, mission elapsed time, clock synchronization • Payload to orbiter - payload mounted sensor attitude information • Pointing accuracy - $\pm$ 0.5 deg; $\pm$ 0.01 deg per second Cooperative Target / Passive Target • Rendezvous - Range: 300 n mi to 100 ft / 24 n mi to 100 ft Range Rate: 1476 ft/sec to 0 / 492 ft/sec to 0
Attitude Control System	• Pointing capability - use RCS to meet pointing accuracy as defined above.
Tracking, Telemetry, and Command	• Telemetry - attached payloads-to-orbiter - 25k bps hardline data to orbiter - 256k bps data relayed to ground via wideband FM transmitter - released payloads-to-orbiter - S-band, phase modulation 16k bps TM only (unmanned payloads) - C&W interface provided from payload to orbiter. • Commands - orbiter-to-attached payloads - hardline, 2k bps, generated on board orbiter or relayed from ground, encoded and interleaved to provide total command rate of 9k bps - orbiter-to-released payloads - same rate as attached payloads, S-band, phase modulation • Tracking - orbiter has the capability to track cooperative targets up to 300 n mi and passive targets up to 12 n mi. • Telemetry and command range - 30 n mi • Video - orbiter provides capability to transmit video data time-shared with wideband payload data
Electrical Power Systems	• H_2/O_2 fuel cell power plants • Can supply 1 kW average and 1.5 kW peak to payloads during *all* mission phases • Can supply 5 kW average and 8 kW peak to payload during *most* orbital operations mission phase • 50 kWH energy allocated to payloads • 28 vdc nominal, two wire, structure ground

	released payloads to orbiter - S-band, phase modulation 16k bps TM only (unmanned payloads) - C&W interface provided from payload to orbiter. ● Commands - orbiter-to-attached payloads - hardline, 2k bps, generated on board orbiter or relayed from ground, encoded and interleaved to provide total command rate of 9k bps - orbiter-to-released payloads - same rate as attached payloads, S-band, phase modulation ● Tracking - orbiter has the capability to track cooperative targets up to 300 n mi and passive targets up to 12 n mi. ● Telemetry and command range - 30 n mi ● Video - orbiter provides capability to transmit video data time-shared with wideband payload data
Electrical Power Systems	● H_2/O_2 fuel cell power plants ● Can supply 1 kW average and 1.5 kW peak to payloads during *all* mission phases ● Can supply 5 kW average and 8 kW peak to payload during *most* orbital operations mission phase ● 50 kWH energy allocated to payloads ● 28 vdc nominal, two wire, structure ground
Fluids	● Orbiter will provide capability to fill, vent, and drain payload cryogenic propulsion systems with the payload installed in payload bay. ● Payloads with earth-storable propellants shall be loaded before installing payloads in the payload bay.
EVA/IVA	● Consumables loaded for three, 2-man, 4-hour EVAs ● Exterior lighting and interior lighting within orbiter bay provided ● IVA maximum package envelope - 22 x 22 x 50 in. (unsuited) ● EVA maximum package envelope - 18 x 18 x 50 in. (suited) ● 40-in. diameter hatch
Shuttle Remote Manipulator System	● Stowed outside of 15-ft diameter x 60-ft payload bay ● 570 in. (47.5 ft) maximum reach ● 190 in. (15 ft 10 in.) minimum reach ● Capable of deploying or retracting a 32,000 lb payload in less than 7 minutes ● One arm provided charged to orbiter; second arm can be provided, but charged to payload ● Capable of deploying or retrieving multiple ($\leq$ 5) payloads
Thermal Control System	● Average heat rejection capability dedicated to payloads Nominal - 3,400 BTU/hr Peak - 5,200 BTU/hr ● Orbital operations (orbiter electrical power requirements <8kW) heat rejection capability dedicated to payloads can be increased to Nominal - 11,250 BTU/hr Peak - 21,500 BTU/hr ● Thermal attitude constraints ● Payload heat rejection - accomplished by heat exchanger located in Freon 21 loop of ATCS--water is the coolant fluid on the payload side. Flow is 550 lb/hour
Operational	● Shuttle system capable of launch within 24 hours of notification ● Shuttle system capable of launch from standby status within two hours, and to hold in standby status up to 24 hours ● Shuttle system capable of on-pad payload changeout within an interval of 10 hours, until T - 2 hours

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Table III-13 Tug-Maintenance Related Characteristics

Category	Capabilities
Payload Size and Handling	• Tug length - 30 ft; payload bay length remaining - 30 ft. Tug/spacecraft deployment/release and attachment mechanisms included in Tug 30-ft length. • Tug diameter - 176 in. (14 ft, 8 in.) • Tug wet weight is 58,679; orbiter maximum delivery into orbit is 65,000 lb, allowing 6,321 pounds spacecraft into 160 n mi. • Tug payload weight capability from 160 n mi altitude to geosynchronous altitude • For deploy/retrieve of different spacecraft, refer to Chapter XI, Item E-6 Mission — Max. Weight Deploy only — 7,926 lb Retrieve only — 3,396 lb Deploy and retrieve (60% apart in orbit) — 2,070 lb Deploy and retrieve (same location) — ~ 2,500 lb
Guidance and Navigation	• Position and velocity accuracy, ± 2.7 n mi - Position; ± 16.4 fps - velocity • Placement Accuracy — Geosynchronous Orbit / Low Earth Orbit Position — 27 n mi / 5.4 n mi Velocity — 32.8 fps / 32.8 fps • Stationkeeping accuracy: Longitudinal velocity, 0.1 - 10 fps Lateral velocity, 0.5 fps Angular misalignment, ±10 deg Angular rate, ±1 deg/ sec • Spacecraft insertion accuracy: Geosynchronous orbit Semimajor axis, ±20 n mi Inclination, ±0.1 deg Longitude, TBD Low earth orbit TBD • Docking accuracy: Centerline miss distance, 0.1 ft Misalignment angle, 0 - 5 deg Constant velocity, 0.1 - 1.0 fps longitudinal 0 - 0.3 fps lateral 0 - 0.5 deg/s angular
Attitude Control System	• Tug/spacecraft retrieval alignment (same as docking accuracy) • Spacecraft attitude and rate accuracy (following deployment): Translational velocity, 0+ to 5.0 fps Attitude rates, ±0.1 deg/sec Attitude, ±2 deg • Spacecraft pointing on tug - ±0.2 deg • Tug capable of accepting spacecraft spin/despin device; tug will initiate device at spacecraft deploy/retrieve for spin rate ≤ 100 rpm • Tug capable of providing any orientation or attitude required by spacecraft during tug/spacecraft operation
Data/Communications	• Data link between spacecraft and tug by hardline only

	Constant velocity, 0.1 - 1.0 fps longitudinal 0 - 0.3 fps lateral 0 - 0.5 deg/s angular
Attitude Control System	• Tug/spacecraft retrieval alignment (same as docking accuracy) • Spacecraft attitude and rate accuracy (following deployment): Translational velocity, 0+ to 5.0 fps Attitude rates, ±0.1 deg/sec Attitude, ±2 deg • Spacecraft pointing on tug - ±0.2 deg • Tug capable of accepting spacecraft spin/despin device; tug will initiate device at spacecraft deploy/retrieve for spin rate ≤ 100 rpm • Tug capable of providing any orientation or attitude required by spacecraft during tug/spacecraft operation
Data/Communications	• Data link between spacecraft and tug by hardline only • Tug will be active element in spacecraft rendezvous/retrieval • Orbiter will be active element in tug/spacecraft rendezvous • Laser radar for rendezvous and docking Range - maximum 300 n mi nominal 50 n mi minimum 30 n mi • Tug will relay spacecraft data to orbiter for relay while in orbiter bay. • Telemetry data ≈ 200 measurement capability (all tug - no capability of spacecraft TM) • Tug video system included for post-deployment visual inspection of spacecraft • No requirement for orbiter/tug communications following tug deployment until tug/orbiter rendezvous and retrieval operations begin
Electrical Power Systems	• Tug will have two fuel cells - 2 kW average power each; 3.5 kW peak power each • One auxiliary battery - 25 amp hr • Tug shall supply spacecraft 300 to 600 watts from tug deployment until tug retrieval • Tug power requirements estimated to be 967 watts, average • Steady state voltage, 28 vdc, +4.5, -4.0 • Orbiter power available to both tug and spacecraft while in orbiter payload bay
Fluids	• Tug shall provide capability for spacecraft fill, drain, dump, pressurization, and venting via orbiter service panels • No propellant sharing between tug and spacecraft or between tug and orbiter • Only fluid interface between tug and spacecraft limited to meet pressure and fluid dump requirements for orbiter payload bay safety
Thermal Control System	• Tug shall provide interconnects for spacecraft cooling in orbiter bay via cooling provisions of the orbiter • No vehicle orientation constraints imposed by TCS • Tug has both active and passive TCS, including Freon 21 loop, insulation, and heat pipes
Operational	• Tug turnaround time ≈ 258 work hours from orbiter landing • Tug mission duration 154 hours from deployment through retrieval • Tug shall have capability to accept changes in mission assignment (not including spacecraft changeout), target, or spacecraft ephemeris data to within two hours before launch • Tug capable of launch from standby status within two hours • Maximum standby in launch configuration is 24 hours • Baseline tug available for flights in December 1983

FOLDOUT FRAME

Table III-14 Ground Support-Maintenance Related Capabilities

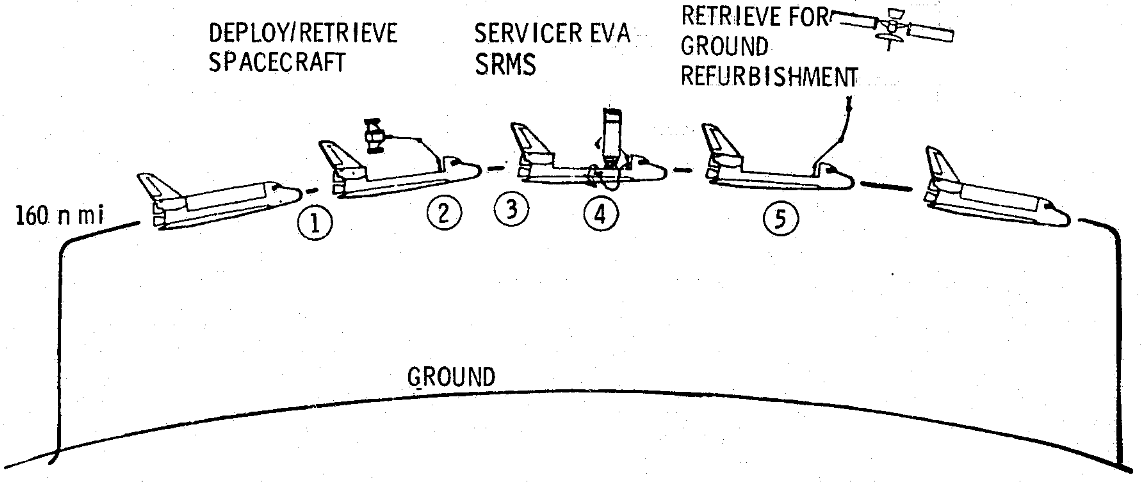
Category	Capabilities
Payload Size and Handling	• Payload removal or installation with the orbiter in the vertical position only at the launch pad • Payload removal or installation with the orbiter in the horizontal position at all other locations • Pad access to payload accomplished through the payload doors or the orbiter crew.compartment • Shuttle system flight vehicle capable of turnaround in less than 160 working hours spanning 14 days • GSE stations involving payload and maintenance - Orbiter maintenance and checkout facility • orbiter and payload safing • payload removal and installation and interface verification • verification of orbiter/payload communication - Launch pad station • prelaunch checkout • payload removal and installation - LRU maintenance station • maintenance, repair, test, analyses, acceptance, and packaging - Launch processing system station
Data/Communications	• Provisions supplied on pad for both RF and hardline (umbilical) interfaces between orbiter and GSE communications for prelaunch voice, TM, video, command checkout
Fluid	• Fluid interfaces to payload provided by GSE
Thermal Control System	• GSE ground thermal conditioning available within 30 minutes after touchdown for payload bay
Operational	• Installation and/or removal of OMS kits without impact to, or by, installed payloads

descriptions of maintenance-related capabilities for the orbiter, tug and ground items, respectively. These types of items were used throughout the study to help assess economic and technical impacts of applying maintenance using elements of the STS.

C. MAINTENANCE MISSION SCENARIOS

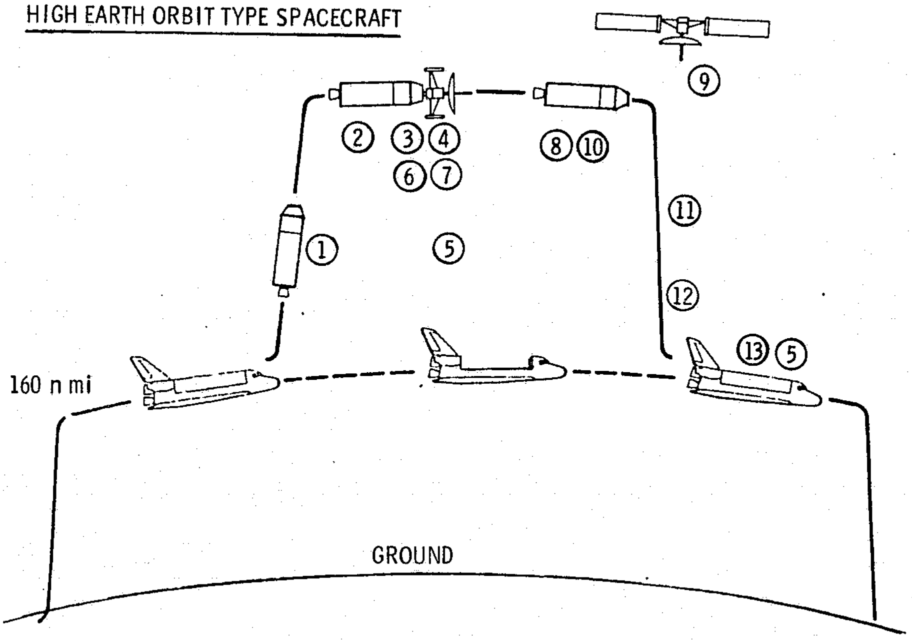
In order to help evaluate maintenance concepts, mission scenarios were prepared for all the maintenance modes. Different scenarios were prepared for LEO missions, for HEO missions, and for certain MEO missions. These scenarios were used to help evaluate compatibilities between maintenance concepts and the STS elements, to help evaluate STS impacts, to obtain rough estimates of mission timelines, and to help determine feasibility of using various maintenance concepts with different spacecraft. Figure III-7 presents the baseline LEO maintenance mission where the shuttle orbiter can reach any spacecraft in LEO and can service it with any of the maintenance concepts of shuttle remote manipulator system (SRMS), EVA, or an on-orbit servicer mechanism. Approximately 159 hours are available for maintenance activities. Figure III-8 presents a baseline HEO maintenance mission, also used on several MEO missions, where the tug takes the maintenance concept, in this case a servicer mechanism, to HEO, services the spacecraft, and then returns to the orbiter. Meanwhile, the orbiter can be conducting pallet experiments while the tug is away, or after it comes back. Approximately 100 hours are available for servicing activities and transfer between spacecraft. The third scenario (Figure III-9) presents an atypical mission, but one which may prove feasible in some cases. In this scenario, the spacecraft is in a medium orbit and one tug can retrieve the spacecraft, bring it down to the orbiter where it is serviced, replace the spacecraft in its correct orbit, and return to the orbiter. There were only 4 programs in the maintenance applicable set where this technique could be used, but it may warrant consideration for these missions. The last scenario (Figure III-10) presents a mission which was considered, but dropped because of its potential high costs, as compared to all the other missions. In this mission, spacecraft in HEO are returned to the orbiter by one tug, serviced, and then replaced in HEO by a second tug which also requires a second orbiter flight. Although technically feasible, the costs of this mode do not warrant further consideration in

LOW EARTH ORBIT TYPE SPACECRAFT



		ESTIMATED TIME EVENT (hr)	ESTIMATED TIME TOTAL (hr)
1.	SHUTTLE LAUNCH, OPEN ORBITER CARGO BAY DOORS AND PERFORM INITIAL PAYLOAD CHECKOUTS	3	3
2.	DEPLOY SPACECRAFT USING SRMS	1	4
3.	PERFORM FINAL SPACECRAFT FUNCTIONAL CHECKOUTS	1	5
4.	ORBITER AND CREW AVAILABLE TO CONDUCT LEO SPACECRAFT MAINTENANCE ACTIVITIES		
	A. ON-ORBIT SERVICER	159	
	B. EVA		
	C. SRMS		
5.	RETRIEVE ONE OR MORE LEO SPACECRAFT FOR EARTH RETURN (PERFORM GROUND REFURBISHMENT)		164

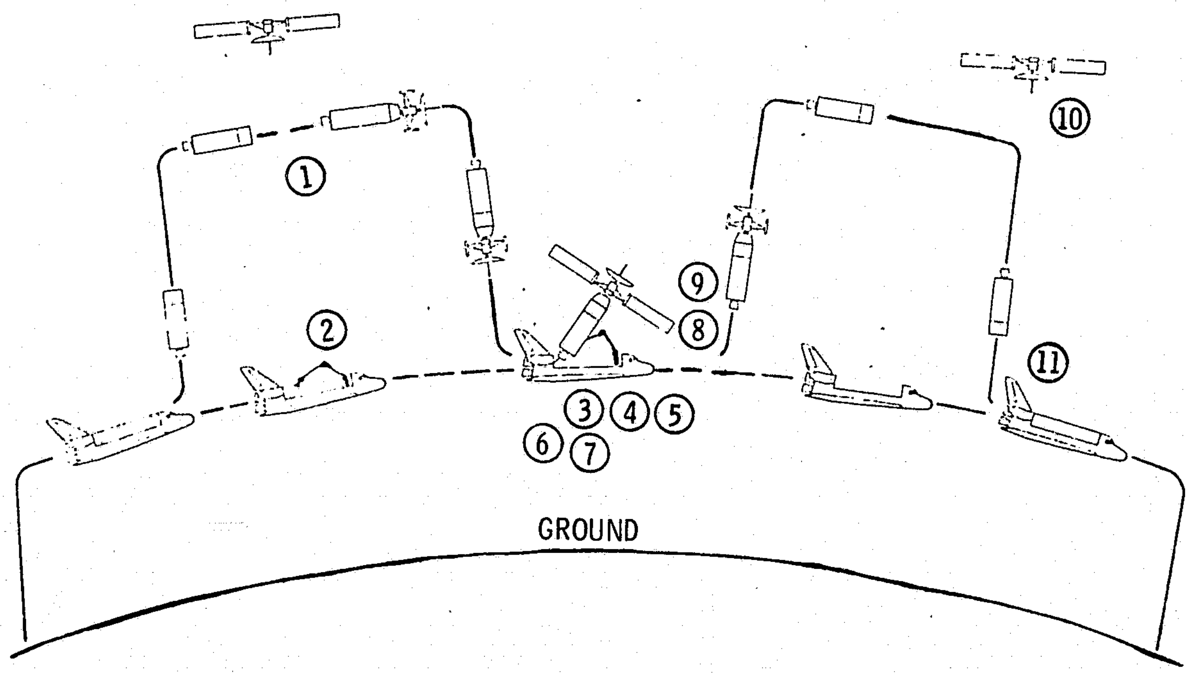
Figure III-7 LEO Maintenance Mission



		ESTIMATED TIME EVENT (hr)	TOTAL (hr)
1.	TUG SEPARATES FROM ORBITER AND TRANSFERS SERVICER TO HEO	22	22
2.	CIRCULARIZE TUG/SERVICER AT GEOSYNCHRONOUS, COAST AND ORBIT TRIM	11	33
3.	TUG/SERVICER RENDEZVOUS AND DOCKS WITH SPACECRAFT	6	39
4.	SPACECRAFT DEACTIVATION	2	41
5.	ORBITER AND CREW AVAILABLE TO CONDUCT PALLET EXPERIMENTS IN LEO~120	--	--
6.	SERVICER PERFORMS MAINTENANCE ACTIVITIES BY PRE-PROGRAMMED DIRECTION AND/OR REMOTE MANNED GROUND CONTROL		
	A. TIME AVAILABLE FOR SERVICING ONE SPACECRAFT ~100	--	141
	B. AVERAGE TIME AVAILABLE FOR SERVICING ONE OF TWO ~40	--	--
	C. AVERAGE TIME AVAILABLE FOR SERVICING ONE OF THREE ~15	--	--
7.	SPACECRAFT ORIENTATION, ACTIVATION AND PRELIMINARY CHECKOUT	1	142
8.	TUG/SERVICER SEPARATION FROM SPACECRAFT	1	143
9.	FINAL SPACECRAFT CHECKOUT	2	145
10.	REPEAT STEPS 3 THROUGH 9 FOR ADDITIONAL SPACECRAFT	--	--
11.	TUG POSITIONED FOR GEOSYNCHRONOUS DEBOOST AND TRANSFER OF SERVICER TO 170 n mi PERIGEE	12	157
12.	TUG CIRCULARIZES AT 170 n mi	3	160
13.	ORBITER RENDEZVOUS AND RETRIEVES TUG/SERVICER TO CARGO BAY	4	164

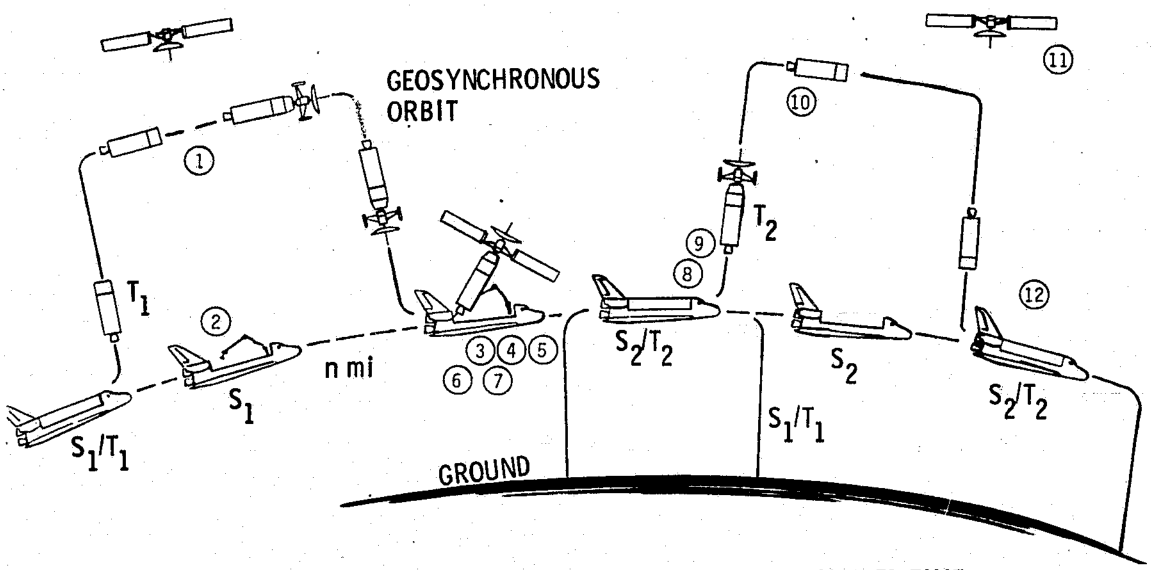
Figure III-8 HEO Maintenance Mission

MEDIUM EARTH ORBIT TYPE SPACECRAFT



		ESTIMATED TIME EVENT (hr)	TOTAL (hr)
1.	TUG SEPARATES FROM ORBITER, TRANSFERS TO, DOCKS WITH AND RETRIEVES MAINTAINABLE SPACECRAFT FROM MEDIUM EARTH ORBIT (MEO)	56	56
2.	ORBITER AND CREW AVAILABLE TO CONDUCT PALLET EXPERIMENTS IN LEO ~50	--	--
3.	SPACECRAFT PLACED IN ORBITER BAY USING SRMS	4	60
4.	SPACECRAFT POSITIONED AND PREPARED FOR MAINTENANCE	2	62
5.	SPACECRAFT MAINTENANCE PERFORMED USING ONE OR MORE OF THE FOLLOWING: SRMS, EVA, AND ON-ORBIT SERVICER ~32	32	94
6.	FUNCTIONAL CHECKS PERFORMED	2	96
7.	TUG/SPACECRAFT SEPARATION FROM ORBITER	1	97
8.	PERFORM SPACECRAFT FUNCTIONAL CHECKOUT	2	99
9.	TRANSFER TUG/SPACECRAFT TO MEDIUM EARTH ORBIT AND DEPLOY SPACECRAFT	37	136
10.	FINAL SPACECRAFT CHECKOUT PERFORMED	2	138
11.	TUG DEPARTS FROM MEO TO 170 n mi PERIGEE, CIRCULARIZES AT 170 n mi AND IS RETRIEVED BY THE ORBITER FOR RETURN TO EARTH	22	160

Figure III-9 MEO Maintenance Mission



		ESTIMATED TIME EVENT (hr)	TOTAL (hr)
1.	TUG SEPARATION FROM ORBITER AND TRANSFERS, DOCKS AND RETRIEVES SPACECRAFT FROM GEOSYNCHRONOUS ORBIT	56	56
2.	ORBITER AND CREW AVAILABLE TO CONDUCT MAINTENANCE ACTIVITIES IN LEO ~ 100	--	--
3.	SPACECRAFT PLACED IN ORBITER BAY USING SRMS	4	60
4.	SPACECRAFT POSITIONED AND PREPARED FOR MAINTENANCE	2	62
5.	SPACECRAFT MAINTENANCE PERFORMED USING ONE OR MORE OF THE FOLLOWING: SRMS, EVA AND ON-ORBIT SERVICER ~100	100	162
6.	SPACECRAFT FUNCTIONAL CHECKS PERFORMED	2	164
7.	SPACECRAFT SEPARATED FROM TUG AND DEPLOYED BY SRMS	2	166
8.	SECOND TUG FROM SECOND ORBITER TRANSFERS TO SPACECRAFT AND DOCKS	4	170
9.	TUG TRANSFERS SPACECRAFT TO GEOSYNCHRONOUS ORBIT AND DEPLOYS	36	206
10.	TUG LOITER TIME OF UP TO 110 hr	90	296
11.	FINAL SPACECRAFT CHECKOUT PERFORMED	2	298
12.	TUG DEPARTS FROM GEOSYNCHRONOUS ORBIT TO 170 n mi PERIGEE, CIRCULARIZES AT 170 n mi AND IS RETRIEVED BY THE ORBITER FOR RETURN TO EARTH	22	320

Figure III-10 HEO Maintenance Mission - Two Tugs

this study. The prime missions considered for this study involved the baseline orbiter and tug missions of Figures III-7 and III-8.

D. FUNCTIONAL AND HARDWARE REQUIREMENTS

In order to perform complete analyses of the maintenance concepts, it was necessary to identify the effects of the maintenance concepts upon the spacecraft, the STS, and other equipment. It was also necessary to know the requirements of the maintenance concepts upon the systems that will actually perform the maintenance. The work performed in this part of the study investigated the service mission phases for identified maintenance concepts and generated the requirements of the maintenance concept upon the servicing system, the STS (orbiter, tug, and ground support) and the spacecraft.

This was performed by first identifying the maintenance concepts (see below) and by then determining all the service mission phases for each concept. Twenty-four separate phases were identified for all of

I	Ground refurbishment using the Orbiter
II	Ground refurbishment using the Tug, or the Earth Orbital Teleoperator System (EOTS)
III	Built-On
IV	Self-repair
V	IVA
VI	EVA
VII	Shuttle Remote Manipulator System
VIII	On-Orbit Maintenance from the Orbiter
IX	On-Orbit Maintenance from the Tug or EOTS

the concepts. The most phases employed in any one concept were 21. Table III-15 presents the maintenance mission phases.

Functional requirements were then generated for each mission phase to account for all concepts. Each separate functional requirement was then reviewed to determine the hardware or other cost requirements that it would impose on the servicing system, the elements of the STS, and the spacecraft. To eliminate redundant work, requirements that were the same for more than one concept were so identified. Table III-16 presents a typical example of the data generated on this task.

Table III-15 Maintenance Mission Phases

Servicer Mission Phases	Applicability								
	I	II	III	IV	V	VI	VII	VIII	IX
Premission	X	X	X	X	X	X	X	X	X
Mission Preparation	X	X			X	X	X	X	X
Prelaunch	X	X			X	X	X	X	X
Orbiter Launch/Ascent	X	X			X	X	X	X	X
Orbiter Orbital Operations	X	X			X	X	X	X	X
Tug or EOTS/Servicer Checkout		X							X
Tug or EOTS Deployment		X							X
Tug or EOTS Orbit Transfer		X							X
Tug or EOTS Orbital Operations		X							X
Servicer Checkout	X	X	X	X			X	X	X
Servicer Deployment (Preparation)	X	X	X	X			X	X	X
Spacecraft Preparation for Servicing	X	X	X	X	X	X	X	X	X
Rendezvous	X	X			X	X	X	X	X
Docking (Attachment)	X	X			X	X	X	X	X
Pre-EVA (IVA)					X	X			
EVA (IVA) Operations					X	X			
Servicing Operations			X	X	X	X	X	X	X
Spacecraft Checkout	X	X	X	X	X	X	X	X	X
Post-EVA (IVA)					X	X			
Undocking (Release) of Spacecraft	X	X			X	X	X	X	X
Spacecraft Preparation for Nominal Operation	X	X	X	X	X	X	X	X	X
Tug or EOTS Return to Orbiter		X							X
Tug or EOTS/Orbiter Rendezvous and Mating		X							X
Orbiter Preparation, Reentry and Landing	X	X			X	X	X	X	X
Post-Mission	X	X			X	X	X	X	X

The data gathered under this task was used in the other tasks of this program to help analyze concepts, to help establish costs, to aid in the setting-up of the work breakdown structure (WBS) and to help establish timeline sequences.

Table III-16 Functional and Cost-Generating Requirements Table, Typical

	ON-ORBIT SERVICING FUNCTIONAL REQUIREMENTS														
	HARDWARE AND OTHER COST GENERATING REQUIREMENTS						APPLICABILITY								
FUNCTIONAL REQUIREMENTS	SERVICER	ORBITER	TUG OR EOTS	SPACECRAFT	GND SUPT	OTHER	I	II	III	IV	V	VI	VII	VIII	IX
PROVIDE POWER TO SERVICER FROM CARRIER VEHICLE	HARDWIRED ELECTRICAL POWER TRANSMISSION EQUIPMENT, CORRECTORS						X	X					X	X	X
		HARDWIRED ELECTRICAL POWER TRANSMISSION INTERFACE CONNECTORS					X							X	
			HARDWIRED ELECTRICAL POWER TRANSMISSION INTERFACE CONNECTORS						X						X
TRANSMIT AND RECEIVE TELEMETRY SIGNALS TO & FROM CARRIER VEHICLE	SENSORS, DATA MANAGEMENT EQUIPMENT, HARDWIRED INTERFACE CONNECTORS						X	X	X	X			X	X	X
		DATA RECEPTION & TRANSMISSION EQUIPMENT FOR SERVICER DATA, HARDWIRED INTERFACE CONNECTORS						X						X	
			DATA RECEPTION & TRANSMISSION EQUIPMENT FOR SERVICER DATA, HARDWIRED INTERFACE CONNECTORS						X						X

IV. MAINTENANCE CONCEPTS

The selection of a maintenance concept is a primary study output and has received significant attention during the course of the study. The evaluations were made in a series of steps successively reducing the number of candidate systems so the selected ones could be addressed to a higher level of detail.

The maintenance concepts evaluated in this chapter are shown in Figure IV-1. The major objective is to examine the concepts for technical feasibility and to provide a level of concept definition which is compatible with the needs for performing spacecraft interface analysis (Chapter V), STS impact analysis (Chapter VIII), cost generation and analysis (Chapter IX), and to select a recommended system (Chapter X).

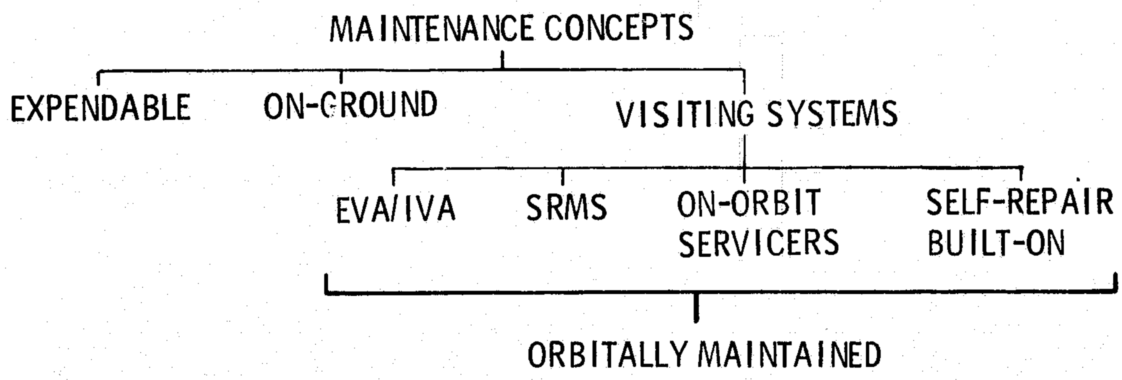


Figure IV-1 Maintenance Approaches

Since there are such a large number of maintenance concepts, a realistic and valid comparison within limited resources is difficult. However, an effective time-saving approach was evolved which permitted a valid comparison to be made. The approach involves two fundamental factors. First a critical grouping of similar concepts is performed at each of the concept levels. Then one concept of the group is chosen and carried forward in the analyses as representing the total group. The second important factor relates to working each level of comparison to a proper degree of detail. The degree of detail should be adequate to draw the conclusions required, but it should not go deeper and waste project time.

The categorization system of Table IV-1 was developed so that greater confidence in the identification of all maintenance concepts could be

Table IV-1 Maintenance Concept Categories

• GROUND REFURBISHABLE
• SELF CONTAINED
BUILT-ON
SELF-REPAIR
• MANNED
IVA
EVA
• SHUTTLE REMOTE MANIPULATOR SYSTEM
• ON-ORBIT SERVICER

obtained. It was important that all alternative maintenance concepts be identified and compared. The concept categorization is sufficiently general that any single concept should fit into the categorization and thus have been effectively evaluated in this study. It provided a method for grouping similar concepts by system characteristics such as operational utility, functional capability and hardware utilization. It also permits the grouping of similar concepts together so that a single concept can be used to represent the group. This approach will permit evaluation of all concepts to a greater depth than would be possible individually. The three basic advantages of this approach include the capability to represent similar concepts by a single concept, provide a tracing capability for every concept from its original source to its eventual disposition with supporting rationale, and provide a flow structure by which new concepts identified in later tasks may be grouped, compared, and evaluated on a consistent basis.

Note that the categorization system does not relate concepts to the elements of the transportation system (orbiter, tug, or free flyer). These effects are expressed in terms of STS impacts. Note also that the

control mode (automatic, remotely manned, manned) is not included in the categorization. Man's direct presence is only involved in the on-orbit operations associated with IVA and EVA. The shuttle remote manipulator system is nominally to be operated in a remotely manned mode. All the other maintenance concepts, including the deployment and retrieval activities associated with ground refurbishment, can be operated automatically or remotely by man.

Our approach is illustrated in the subtask interrelationships and flow shown in Figure IV-2. In Section A a top level look at the complete field

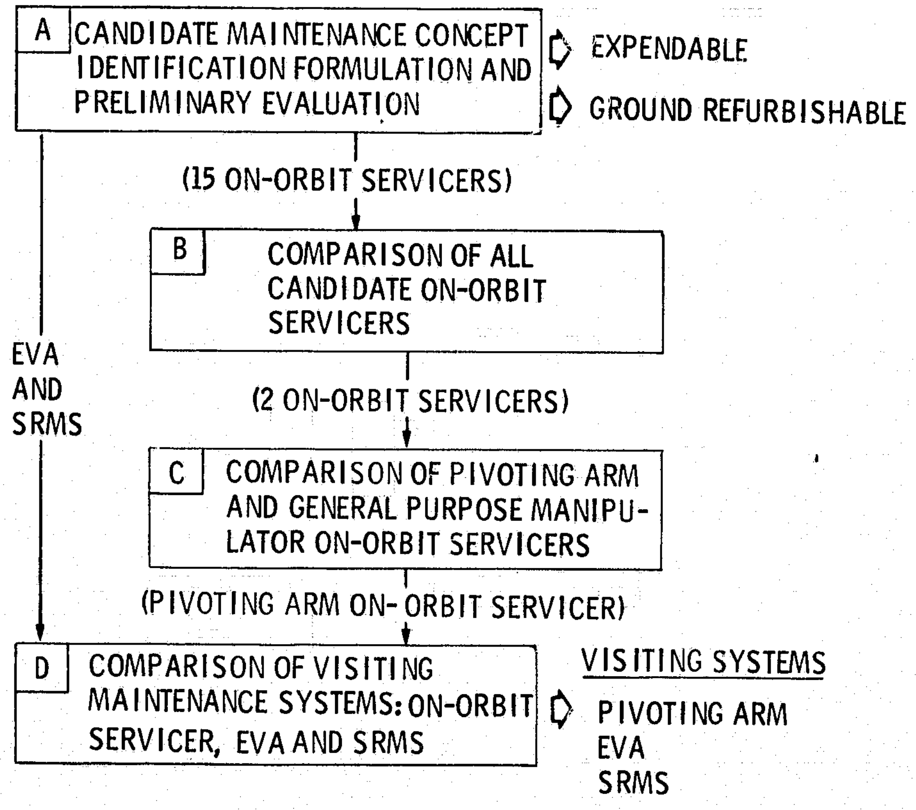


Figure IV-2 Maintenance Concept Evaluation Flow

of candidate maintenance concepts is performed. There are four outputs of this section. The expendable and ground refurbishable modes are defined for use in other tasks. Fifteen on-orbit servicer conceptual designs are identified for a categorizing and comparative evaluation in section B which results in the selection of two on-orbit servicers: pivoting arm (TRW), and general purpose manipulator (MDAC). These two servicers are evaluated to a greater level of detail (Section C), where the pivoting arm is recommended as the

best on-orbit servicer and is recommended to represent the total field of on-orbit servicers in further evaluations.

Finally, in section D the pivoting arm on-orbit servicer, EVA and SRMS maintenance concepts are compared for operation in low earth orbit. This last section concludes that these three maintenance concepts are all technically feasible. Advantages and disadvantages of each are compared. Several significant equipment areas are identified which can affect cost comparisons performed in Chapter IX. The significant equipment impact areas are:

(1) Spacecraft man-rating for EVA,
(2) Additional support structure for large spacecraft for EVA and SRMS maintenance (weight, stowage and operating volumes), and
(3) Addition of a module exchange capability to the SRMS (positional accuracy and manipulator degrees of freedom).

A. CANDIDATE MAINTENANCE CONCEPT IDENTIFICATION AND PRELIMINARY EVALUATION

The candidate maintenance concepts are shown in Figure IV-1. The concepts listed represent the totality covered in the literature with the exception that fifteen on-orbit servicers (Table IV-2) have been identified.

Table IV-2 Field of On-Orbit Servicers

1.	MDAC DIRECT ACCESS
2.	AEROSPACE CORPORATION
3.	BELL AEROSPACE CARTESIAN COORDINATE
4.	RI UOP A (EXTERNAL)
5.	PIVOTING ARM TYPE
	A. RI UOP B (INTERNAL)
	B. MSFC
	C. TRW
	D. BELL AEROSPACE CYLINDRICAL COORDINATE
6.	GENERAL PURPOSE MANIPULATOR TYPE
	A. RI GEOSYNCHRONOUS PLATFORM
	B. MDAC EXTERNAL
	C. MMC GENERAL PURPOSE
	D. GE AGOES BOOM
7.	SHUTTLE CARGO BAY ONLY
	A. MSFC SHUTTLE MODULE EXCHANGE
	B. RI EOS
	C. SPAR/DSMA EOS

A description of each servicer and a top level evaluation are contained in the next section. As the table shows, a first grouping of the servicers was performed to aid in the evaluation process. This grouping is reviewed in the next section as more servicer details are introduced.

The expendable and ground refurbishable maintenance modes require very little technical definition. The methods of design and on-orbit functioning of expendable spacecraft has been established and verified over a considerable number of years. The ground refurbishable maintenance mode basically requires the addition of retrieval capability to an expendable spacecraft. The advantages and disadvantages for ground refurbishable maintenance are listed in Table IV-3. There are not any technical feasibility questions. Both the expendable and ground refurbishable maintenance modes are investigated for cost impacts in Chapter IX.

Table IV-3 Ground Refurbishment Maintenance Concept

DESCRIPTION
THIS CONCEPT UTILIZES THE STS TO RETRIEVE AND RETURN TO EARTH THE SPACECRAFT FOR A COMPLETE REFURBISHMENT OR MAINTENANCE ACTIVITIES.
ADVANTAGES
MINIMUM MAINTENANCE MODIFICATION REQUIRED TO SPACECRAFT DESIGN. THE SPACECRAFT CAN BE REFURBISHED TO A LIKE-NEW CONDITION. THERE IS NO NEW GSE DEVELOPMENT FOR GROUND MAINTENANCE. NO REQUIREMENT FOR SPECIAL MODULE MOUNTING; FAULT DETECTION, ISOLATION, AND VERIFICATION EQUIPMENT; SERVICING MECHANISMS; RESTRAINTS SYSTEMS, ETC. CAPABILITY TO REPAIR/REFURBISH AND CHECKOUT OF EQUIPMENT THAT CANNOT BE DONE ON-ORBIT. UPDATE SPACECRAFT CAPABILITIES.
DISADVANTAGES
SPACECRAFT MUST BE COMPATIBLE WITH RETRIEVAL OPERATION. TWO SHUTTLE LAUNCHES TO ACCOMPLISH MAINTENANCE TASK, ONE TO RETRIVE AND ONE TO RELAUNCH SPACECRAFT. MAINTAIN A REPAIR FACILITY ON THE GROUND WITH ALL THE SPECIAL HANDLING EQUIPMENT AND REPLACEABLE MODULES/COMPONENTS. DOWNTIME IS GREATER FOR THIS CONCEPT THAN ANY OTHER CONCEPT. EQUIPMENT THAT IS NOT REPLACED MUST GO THROUGH A RE-ENTRY AND ANOTHER LAUNCH ENVIRONMENT WHICH PLACES ADDITIONAL STRESS ON THE EQUIPMENT. EQUIPMENT WHICH IS DEPLOYED FOR OPERATION REQUIRES EITHER A RETRACTION OR DROP-OFF CAPABILITY PRIOR TO EARTH RETURN.
CONCLUSIONS
THE ADVANTAGES OFFERED BY THIS CONCEPT PROVIDE A HIGHER LEVEL OF REFURBISHMENT THAN ANY OTHER MAINTENANCE CONCEPT REVIEWED, HOWEVER THE DISADVANTAGES ARE ALSO COSTLY. THEREFORE, THIS CONCEPT SHOULD BE FURTHER INVESTIGATED FOR SPECIAL CASES.

The results of our investigation of built-on and self-repair maintenance concepts are summarized in Tables IV-4 and IV-5, respectively. The built-on concept maintains a spacecraft in an acceptable operational

Table IV-4 Built-On Maintenance Concept

DESCRIPTION
THIS CONCEPT WILL MAINTAIN A SPACECRAFT IN AN ACCEPTABLE OPERATIONAL CONDITION FOR THE DURATION OF A MISSION WITHOUT PHYSICAL ASSISTANCE FROM ANY OTHER SOURCE SUCH AS THE STS BY USE OF A BUILT-ON SERVICING MECHANISM AND THE EXCHANGE MODULES STORED IN THE SERVICER.
ADVANTAGES
NEAR CONTINUOUS OPERATIONAL CAPABILITY. NO ADDITIONAL SHUTTLE LAUNCHES OR DOCKING PROVISIONS NEEDED EXCEPT WHERE SPACECRAFT RETRIEVAL IS REQUIRED.
DISADVANTAGES
COST OF A SEPARATE SERVICING MECHANISM FOR EACH SPACECRAFT INCLUDING FAULT DETECTION ISOLATION, AND VERIFICATION EQUIPMENT OF TELEMETRY FOR GROUND DIAGNOSIS AND VERIFICATION. CANNOT BE UPDATED WITH NEW EQUIPMENT. HIGH RISK FACTOR DUE TO THE COMPLEXITY OF THE MECHANICAL EQUIPMENT REQUIRED TO PERFORM THE MAINTENANCE FUNCTIONS AND STORAGE REQUIREMENTS. REQUIRES TAKING REPLACEMENT MODULES FOR MODULES THAT MAY NOT FAIL. CANNOT COMPENSATE FOR MODULES THAT HAVE DESIGN FAILURES. HIGH COST, WEIGHT, VOLUME, AND POWER.
CONCLUSION
THE SAME GOAL OF THIS CONCEPT CAN BE ATTAINED THROUGH SIMPLE REDUNDANCY WITH A HIGHER DEGREE OF RELIABILITY, AND REDUNDANCY IS CONSIDERED A SPACECRAFT DESIGNERS PROBLEM. EVEN THOUGH ELIMINATION OF ADDITIONAL SHUTTLE LAUNCHES IS A CONSIDERABLE COST SAVINGS, THE DISADVANTAGES ARE OF A MAGNITUDE THAT DOES NOT WARRANT FURTHER INVESTIGATIONS.

condition for the duration of a mission without physical assistance from any other source such as the STS by use of a built-on servicing mechanism and the exchange modules are stored in the servicer. The self-repair concept consists of a built-on module exchange mechanism and the capability to repair the failed module by a remotely manned mechanism within the spacecraft. Many of the disadvantages for both these concepts are similar and are very significant. Each spacecraft has its own servicer mechanism with associated weight, volume, reliability, and cost penalties. The offsetting advantage is no additional servicing launches. It was concluded that this does not offset the disadvantages and that the built-on and self-repair

Table IV-5 Self-Repair Maintenance Concept

DESCRIPTION
THIS CONCEPT CONSISTS OF A BUILT-IN MODULE EXCHANGE MECHANISM AND THE CAPABILITY TO REPAIR THE FAILED MODULE BY A MANNED-REMOTE MECHANISM WITHIN THE SPACECRAFT.
ADVANTAGES
NEAR CONTINUOUS OPERATIONAL CAPABILITY. NO ADDITIONAL SHUTTLE LAUNCHES OR DOCKING PROVISIONS NEEDED EXCEPT WHERE SPACECRAFT RETRIEVAL IS REQUIRED. ONLY ONE REPLACEABLE MODULE OF EACH TYPE REQUIRED IN ADDITION TO THE SPARE COMPONENTS. POTENTIAL FOR MORE DIRECT APPLICATION OF STANDARDIZED COMPONENT HARDWARE. PROVIDE REPAIR FOR THE LEVEL AT WHICH FAILURES HISTORICALLY OCCUR.
DISADVANTAGES
A SEPARATE SERVICING MECHANISM FOR EACH SPACECRAFT INCLUDING THE MODULE EXCHANGER, MODULE REPAIR MECHANISM, AND FAULT DETECTION, ISOLATION, AND VERIFICATION EQUIPMENT OR TELEMETRY FOR GROUND DIAGNOSIS AND VERIFICATION. NO MEANS TO UPDATE WITH NEW EQUIPMENT. HIGH RISK FACTOR DUE TO COMPLEXITY OF THE MECHANICAL EQUIPMENT REQUIRED TO PERFORM MAINTENANCE FUNCTIONS AND STORAGE REQUIREMENTS. NO MEANS TO COMPENSATE FOR DESIGN FAILURES THAT MAY OCCUR. STORAGE SPACE REQUIRED FOR REPLACEMENT MODULES AND COMPONENTS THAT MAY NOT FAIL. HIGH POWER, COST, WEIGHT AND VOLUME REQUIREMENTS TO PERFORM MAINTENANCE FUNCTIONS.
CONCLUSION
EVEN THOUGH THE ELIMINATION OF ADDITIONAL STS LAUNCHES WOULD BE A CONSIDERABLE COST SAVINGS, THE DISADVANTAGES ARE OF A MAGNITUDE THAT DOES NOT WARRANT FURTHER INVESTIGATIONS OF THIS CONCEPT.

maintenance concepts did not warrant further investigation.

The IVA maintenance concept utilizes man within a pressurized compartment to accomplish the task of performing the servicing activities on a visiting basis in a low earth orbit. The level of repair could be at the module or component exchange level depending on the overall concept. This concept most nearly represents a ground maintenance concept on-orbit with man-in-the-loop. It has been demonstrated in previous space missions that man can accomplish tasks in a spatial environment with nearly the same efficiency as on the ground. There are some inconveniences such as providing restraints for the man and tools/equipment, but these are not considered constraints. The largest cost factor is providing the life support systems and the resulting additional launch weight for the spacecraft or the repair

facility in the cargo bay of the orbiter. The most benefit is derived by having man in a position to evaluate the condition and perform the required maintenance, which may be different than what was diagnosed prior to the maintenance launch. The advantages and disadvantages of IVA maintenance are summarized in Table IV-6. Spacecraft design information indicates that this type of system would have limited applicability. Thus, it was concluded that IVA maintenance did not warrant further investigation.

Table IV-6 IVA Maintenance Concept

ADVANTAGES
MAN AT THE SITE HAS BENEFITS THAT COMPARE TO ON THE GROUND REPAIR.
CONFIGURATION AND ORIENTATION ARE NOT AS CRITICAL AS REMOTE OR AUTOMATIC OPERATIONS.
MAN HAS THE ABILITY TO WORK AROUND SITUATIONS WHERE MECHANISMS WOULD FAIL.
MAN PROVIDES GREATER DEXTERITY AND COGNITIVE CAPABILITIES.
MAN IS AVAILABLE ON LOW EARTH ORBIT SHUTTLE FLIGHTS.
DISADVANTAGES
LIMITED APPLICABILITY TO SERVICING THE MANY ANTICIPATED SPACECRAFT.
HIGH COST ASSOCIATED WITH PROVIDING THE LIFE SUPPORT SYSTEMS.
HIGH COST TO PROVIDE RELIABILITY REQUIRED FOR MANNED MISSIONS.
MUST BRING SPACECRAFT INTO A LIFE SUPPORTABLE ENVIRONMENT; PRESSURIZE SPACECRAFT; OR PROVIDE GLOVEBOX DESIGN COMPATIBILITY.

In summary, the expendable and ground-refurbishable concepts were accepted for further evaluation and cost considerations, the built-on and self-repair concepts were not further considered, the IVA concept was deemphasized and blended into EVA, the EVA and SRMS concepts were carried to section D, and the on-orbit servicers are discussed in section B.

B. ON-ORBIT SERVICER EVALUATIONS

This section describes the processes by which the 15 servicer concepts of Table IV-2 were categorized into seven major groups. Subsequently, a selection was made of two servicer concepts for detail evaluation in section C. The two servicer concepts selected for expanded consideration are the pivoting arm and the general purpose manipulator.

Servicing mechanisms evolved under contemporary studies have involved methods of moving a module a few feet from the spacecraft to the servicer stowage rack, the precept being that if the task were so limited, then the device should be much simpler than the versatile general purpose manipulators employed for so many terrestrial applications. The fact is that the modules still must be moved in and out, rotated, indexed and positioned in varying degrees; and the result is the simplest mechanisms still require about four degrees-of-freedom plus end effectors for latching and unlatching. Many of the proposed devices are found to be more complex and heavier than a general purpose manipulator and at the same time are limited in versatility.

Most of the module exchange devices proposed to date are limited in that the working range allows servicer module stowage and spacecraft module location on the end only, outer ring facing aft only, or outer ring facing outboard only. If module location is limited to the center or end of the spacecraft, the spacecraft design may not be greatly compromised. A modular spacecraft allows for good subsystem grouping and thermal control with end located modules. Other spacecraft not yet in the planning stage may not be as easily arranged, particularly if replaceable earth pointing instruments are involved, and must maintain continuous operation.

Thus, the issues involved are highly interrelated but involve characteristics and requirements of spacecraft interfaces, a servicing mechanism, a space replaceable module stowage rack, and carrier vehicle capabilities. It was decided to identify potentially effective candidates among the various servicer concepts which deserve the focus of study resources to conduct equitable and meaningful economic evaluations.

A set of screening criteria has evolved, based on a definition of most of the desirable requirements of spacecraft servicing devices. These criteria, which evolved into design guidelines for servicing economy, were useful for assessing impacts to spacecraft design necessary to realize potential benefits of on-orbit servicing. For this evaluation, however, the screening criteria are employed as a metric to maintain consistency in the competitive technical evaluation of the many servicer mechanisms. These screening criteria have been useful in the process of comparing each proposed spacecraft servicer for technical excellence in its implied design, a necessary initial step to identify the more desirable servicers in view of an eventual application to the entire mission model.

1. Mechanism Evaluation Approach

On-orbit servicers have been evaluated with the objective: to review and analyze all previous studies of on-orbit servicing and to reduce this variety to a manageable number of servicer concepts that retain all valid options.

Of the three major elements of the space-borne equipment (servicer mechanism, storage rack, and programmer), the literature search and preliminary considerations led us to emphasize the servicer mechanism. The literature available had little or no data on the programmer, and the programmer can be readily designed to accommodate any servicing requirements. The third space-borne element, the stowage rack, in general can also be reconfigured to better satisfy the servicer requirements, so it was only considered where there were definite limiting factors. The evaluation approach thus emphasizes the alternative servicer mechanisms.

As this evaluation was the first step in a larger evaluation, we carefully examined a broad set of evaluation criteria and selected those which are most applicable to the level of detail available for use, and to a first evaluation.

The servicer mechanism evaluation approach followed the flow shown in Figure IV-3 leading to the evaluation and selection of those mechanisms to be further studied. Fifteen servicer mechanisms were identified in the literature which were reduced to seven by the similarity discussions. Over 85 references were reviewed for data applicable to this study. Some of these

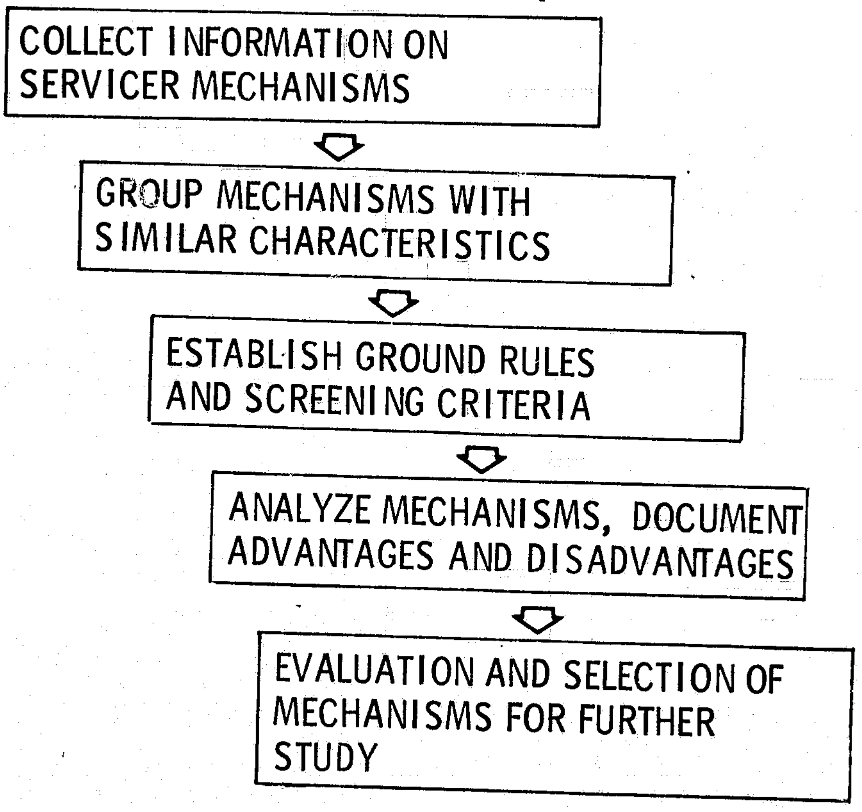


Figure IV-3 Servicer Mechanism Evaluation Flow

provided detailed data on servicer mechanisms. The seven studies of Table II-1 were reviewed for broader considerations, the servicer evaluation criteria, candidate functional guidelines, and candidate design guidelines. These factors are introduced in the paragraphs below. The seven studies are the most recent work for the important continuing servicing studies and present their current conclusions. The data and results of the prior studies were used to identify direction, and provide data and methodology for this study. The various items were analyzed, cross-checked between studies, and verified before being incorporated in the study conclusions.

2. Servicer Mechanism Data Sources

This section identifies the data sources used to obtain data on the various servicer mechanisms proposed in the literature and groups similar mechanisms together.

A formal NASA literature search and a Defense Documentation Center report bibliography provided few useful references. However, a 1973 Navy

Space Systems Activity conference report and the 1973 Proceedings of the Second Conference on Payload Interfaces supported by McDonnell-Douglas Astronautics Company led to most of the active organizations and their orbital maintenance concepts. A partial data deficiency was uncovered at the first quarterly review. It was found that a number of additional references with regard to the GSFC module exchanger mechanism for the earth observation satellite were available.

Over 85 useful references were located. From this, 15 servicer mechanisms were identified. Generally, the references discussed the mechanism and the stowage rack. There was little information on the rest of the maintenance concept; e.g., space-borne programmer, logistics computer, maintenance center, interfaces with the user community, or module refurbishment philosophy. Several of the references included evaluations of four to eight servicer mechanisms. These references identified the criteria used for the comparative evaluations. Other references listed the advantages of particular mechanisms. These advantages were considered as possible screening criteria. The variety of mechanism designs and the differences in evaluation criteria uncovered imply that the design of a module exchanger to meet the requirements of the total maintenance applicable mission model is difficult.

The fifteen servicer mechanisms identified are listed in Table IV-7 by reference to the originating organization with modifiers as necessary. The primary references are indicated in parenthesis in the table as item numbers from Chapter XI. In several cases, supplementary data from other sources were also used.

Several organizations have suggested more than one servicer mechanism. The mechanisms have also been grouped where the alternative concepts are functionally similar. In this way, the number of concepts have been reduced to permit greater depth of detail in the next level of evaluation with no loss of valid approaches. This approach extends the maintenance concept categorization technique to the servicer mechanism level.

Table IV-7 Servicer Mechanisms

1.	MDAC DIRECT ACCESS (K-8)
2.	AEROSPACE CORPORATION (F-19)
3.	BELL AEROSPACE CARTESIAN COORDINATE (K-1)
4.	RI UOP A (EXTERNAL) (M-13)
5.	PIVOTING ARM TYPE
	A. RI UOP B (INTERNAL) (M-13)
	B. MSFC (E-4)
	C. TRW (N-14)
	D. BELL AEROSPACE CYLINDRICAL COORDINATE *
6.	GENERAL PURPOSE MANIPULATOR TYPE
	A. RI GEOSYNCHRONOUS PLATFORM (M-6)
	B. MDAC EXTERNAL (K-2)
	C. MMC GENERAL PURPOSE (J-2)
	D. GE AGOES BOOM (K-1)
7.	SHUTTLE CARGO BAY ONLY
	A. MSFC SHUTTLE MODULE EXCHANGE (K-1)
	B. RI EOS (K-1)
	C. SPAR/DSMA EOS (0-1)

*INFORMAL COMMUNICATION FROM DR. G. GORDON OF COMSAT LABS, SEPTEMBER 1974.

3. Ground Rules For Screening Servicer Mechanisms

This paragraph defines the ground rules used during the screening of the servicer mechanisms and presents the rationale for the ground rules selected. Table IV-8 is a summary of the important ground rules.

Table IV-8 Ground Rules for Screening of Servicer Mechanisms

SPACECRAFT DESIGNED TO BE SERVICEABLE
MODULE EXCHANGE ONLY
ALL MODULES ARE LOCATED ON ONE OR TWO SEPARATE DOCKING FACES OR IN ONE OR TWO ADJACENT TIERS
LARGE ANTENNAS AND SOLAR PANELS ARE ASSUMED TO HAVE LONG LIFE AND HIGH RELIABILITY AND THEREFORE DO NOT NEED REPLACING

Mechanisms would only be evaluated against spacecraft that were designed to be serviceable. This ground rule was suggested by the study Request for Proposal and is consistent with the approach of all of the servicer mechanism references. The primary on-orbit servicing activity would be module exchange. This is consistent with all of the references. All modules are assumed to look alike as far as the servicer mechanism is concerned so that different end effectors need not be considered at this point.

Each maintainable spacecraft can be configured so that all modules can be located on one or two separate docking faces or in one or two adjacent tiers. The ground rule is consistent with the results of all of the reference documentation. For the seven prior studies referenced above, all of them stated that it was definitely possible to configure an on-orbit maintainable version of their reference spacecraft to the level of definition of their study.

Large antennas and solar panels are assumed to have long life and high reliability and, therefore do not need replacing. Again this rule is consistent with the referenced studies. If an antenna has a moveable feed or if some of its components, to be located in the antenna feed area, are not reliable enough, then they can be designed as a space replaceable module. The solar panels are usually sized to meet the lifetime and reliability requirements and need not be replaced. There may be a need for replacement of solar panel drive motors and concepts for their on-orbit replacement have been advanced.

The original approach was to identify a set of servicer requirements from an evaluation of the maintenance applicable spacecraft. Elements were typically to be: functional requirements as in Table IV-9, number of modules, module size, module weight, module location, applicability to many carrier vehicles, and ability to service multiple spacecraft. However, a preliminary review of the servicer concepts showed that many of the mechanisms would be needlessly rejected if they were strictly compared to a set of performance requirements. Thus, this approach was adjusted so that it did not unduly penalize servicer concepts that had been designed for a limited set of spacecraft or for a particular set of ground rules. Recognition was given to the fact that most of the concepts can be reconfigured or rescaled or have functions added to meet given performance requirements. From the original set

Table IV-9 Representative On-Orbit Servicer Functional Requirements

SERVICER SYSTEM FUNCTIONS:

- PROVIDE POWER TO SERVICER FROM CARRIER VEHICLE.
- TRANSMIT AND RECEIVE TELEMETRY SIGNALS TO AND FROM CARRIER VEHICLE.
- TRANSFER STRUCTURAL LOADS FROM SPACECRAFT TO CARRIER VEHICLE.
- THERMAL CONTROL OF MODULES.
- DO NOT INHIBIT OPERATION OF SPACECRAFT OR CARRIER VEHICLE THERMAL CONTROL SYSTEM.
- DO NOT INHIBIT OPERATION OF SPACECRAFT MISSION WHEN SERVICING IS COMPLETE

PROGRAMMER FUNCTIONS:

- COMMAND MECHANISM TO POSITION AT ANY FAILED OR REPLACEMENT MODULE.
- AUTOMATIC SEQUENCING OF MODULE EXCHANGE.
- PROVIDE OBSTACLE AVOIDANCE (ANTENNAS, ARRAYS, ETC).
- MONITOR SERVICING PROCESS.
- ACCEPT AND EXECUTE COMMAND SIGNALS FROM CARRIER VEHICLE.

MECHANISM FUNCTION:

- PROVIDE CAPABILITY TO REACH ALL MODULES.
- ATTACH TO AND RELEASE ALL MODULES.
- TRANSPORT MODULES TO DESIRED LOCATIONS.
- PLACE AND WITHDRAW ALL MODULES.
- PROVIDE TEMPORARY STOWAGE WHILE EXCHANGING.
- OPERATE MODULE LATCHES.

STOWAGE RACK FUNCTIONS:

- PROVIDE MODULE STOWAGE (ALL MODULES FOR ONE OR MORE SPACECRAFT).
- OPERATE MODULE LATCHES (ALTERNATE APPROACH).
- PROVIDE INDICATION OF MECHANICAL LATCHING/ UNLATCHING.

POSSIBLE BACKUP FUNCTIONS:

- REDUNDANT MODULE ATTACHMENT AND RELEASE.
- MANUAL SEQUENCING OF MODULE EXCHANGE.

of servicer performance requirements, we set to one side those which could be reasonably accommodated by servicer design. The others were incorporated into the screening criteria of the next section. The servicer mechanisms were then compared against each other for these criteria.

The following aspects were believed to be applicable to most servicer concepts and could be evaluated independently of the servicer mechanism: programmers, control systems, electrical interfaces, end effectors, attachments, and backup modes. Emphasis in the evaluation was on a comparison of the servicer mechanism used to move the modules.

4. Mechanism Screening Criteria

This paragraph develops the criteria to be used in the top level screening of the servicer mechanisms. The evaluation factors established in this study and from a literature review were combined into Table IV-10. It lists the factors selected for the top level screening. The final concept might well be the one which achieves the best balance between maximum simplicity and maximum versatility. Certainly for a much used and important system

Table IV-10 Factors for Servicer Mechanism Screening

MISSION OPERATIONAL FACTORS
MECHANISM SHOULD BE USABLE ON MANY CARRIER VEHICLES CLASS OF SPACECRAFT PROGRAMS TO WHICH MECHANISM APPLIES. MULTIPLE SPACECRAFT MAINTENANCE.
MECHANISM OPERATIONAL FACTORS
OPERATIONS TIME. SPACECRAFT MULTIPLE FACE ACCESSIBILITY. CONSTRAINTS ON NUMBER, SIZE, LOCATION, AND ORDER OF EXCHANGE OF MODULES. DOES MECHANSIM OPERATE LATCHES. PROVISION FOR TEMPORARY MODULE STORAGE.
MECHANISM CHARACTERISTICS
WEIGHT (MECHANISM, RACK, DOCKING DEVICE, ADAPTER). LENGTH (MINIMUM FOR CARGO BAY STOWAGE). POWER REQUIRED. ANCILLARY EQUIPMENT REQUIRED. VOLUME. COMPLEXITY (I.E., UNRELIABILITY). REQUIRES ADVANCEMENT OF STATE-OF-THE-ART. EFFECTS OF STRUCTURAL FLEXIBILITY. EXCHANGE FORCE REQUIRED.

such as the servicer mechanism, especially in geosynchronous orbit, having a simple system with inherent high reliability is highly desirable. Yet the mechanism must not be too restrictive on the spacecraft designer. The spacecraft designer is faced with a great many important decisions and tradeoffs. To add unnecessary constraints in terms of number of modules, module size, module shape, and module locations could result in excessive spacecraft costs. Thus the approach of reducing constraints on the spacecraft designer is used as a primary criteria in servicer mechanism screening. The selected criteria were grouped under four general headings: versatility, simplicity, length, and weight. These were further defined by the subheadings shown in Table IV-11. The subheadings of Table IV-11 were used to organize the evaluations of each of the 15 servicer mechanisms and are described in detail below. Each of the screening criteria of Table IV-10 appears under one of the Table IV-11 subheadings. Each module exchange mechanism was evaluated on the basis of the configuration described in the reference literature. A "comments" paragraph was included at the end of each mechanism discussion to indicate options to the basic configuration that would make it more useful.

Table IV-11 Servicer Mechanism Screening Criteria

VERSATILITY
VERSATILITY
DOCKING MECHANISMS
SIMPLICITY
MECHANICAL ADVANTAGE
STRUCTURAL FLEXIBILITY
RELIABILITY
LENGTH
SIZE
WEIGHT
WEIGHT

Versatility - Versatility is used to obtain a measure of how few constraints are placed on the spacecraft designer. Under this heading are discussed: module size range, module shape range, numbers of modules, module locations on the spacecraft, applicability to the orbiter, tug and earth orbital teleoperator system (EOTS), ability to service multiple spacecraft, and restrictions on order of exchanging modules. In the evaluations that follow, the desirable values for these items are those which provide the spacecraft designer with the widest choice of how he can design his spacecraft. Heavy restrictions on the spacecraft designer mean that servicing will not be used or that its use will be limited. Module weight is not included as a criteria as servicers can be designed to handle the masses involved, and weight restrictions are more related to carrier vehicle capability.

Mechanical Advantage - Mechanical advantage was used to discuss the geometric/kinematic nature of the mechanism as well as its relative efficiency. Under this heading are discussed: power conversion for each mechanism motion, linear versus rotary motion, latching and unlatching forces, temporary module stowage, and operations time. A small number of devices to convert energy from the electrical to mechanical form is good. Rotary motions are preferred over linear because linear systems tend to be heavier and less reliable. Latching and unlatching forces should be low and conical tapers should be such that the latching force does not result in too high of an unlatching force. A method of temporarily, or permanently, stowing the failed modules removed from the spacecraft must be provided while the replacement modules are installed. Operations time to exchange modules should be a small part of the seven day tug operations time, say one or two hours. The mechanical advantage factors were considered more for their effect on other aspects such as reliability, weight, and operational capability rather than for themselves per se.

Docking Mechanisms - The docking mechanisms were not evaluated; rather the effect of the servicer mechanism on the utilization of a docking mechanism is addressed. Two classes of docking mechanisms were treated: 1) center located, such as the Apollo docking probe and drogue, and 2) peripherally located, such as the baseline tug docking ring or the Apollo/Soyuz docking mechanism. It is desired that the servicer mechanism be adaptable to either

the center or peripheral docking systems so that the spacecraft designer is not limited. The structural load paths during docking and in the docked configuration were considered for their effect on weight and servicer length, both of which should be kept low. Any significant interactions, such as the need to retract the center docking probe in the Bell Aerospace cartesian coordinate system are also discussed. Again, minimum weight and maximum reliability are desired.

Structural Flexibility - Two aspects of structural flexibility were considered: 1) the stiffness of the docked spacecraft, mechanism, and carrier vehicle assembly, and 2) the exchanger mechanism stiffness during module handling. The relative structural stiffness in the docked configuration was evaluated for its effects on weight and attitude control system interactions. Low weight and low interaction effects are desirable. The exchanger mechanism load paths and stiffness during module insertion and latching were evaluated to see if the mechanism is soft enough in the right places for self-alignment and stiff enough in the right places to avoid binding. Several of the mechanisms tend to use redundant guides during module insertion which can cause binding. Load path length during latching was also of concern. As the total load path must take the latching loads, it should be kept short so that undue flexing will not occur and so that weight will be low.

Size - Generally size should be kept low as large size tends to imply high weight or soft structure. The operating configuration was evaluated with regard to the space required to maneuver modules from the stowage rack to the spacecraft and vice versa. A short length is desirable. More important is the size of the servicer when configured for use on the tug and stowed in the orbiter cargo bay. The PUT study showed a significant impact on launch cost sharing benefits if the stowed servicer length was not kept short. Length values were expressed as multiples of module length. The orbiter cargo bay diameter was used as a limiting factor for servicer stowage. As all of the mechanisms were designed with this limitation in mind, it was not a significant discriminator.

Weight - The space-borne weight of the servicer system should be kept low when it is to be carried by the tug to geosynchronous orbit because of the limited tug round-trip capability. Absolute weights were not determined, rather weight contributing factors and relative weight assessments were used.

Reliability - Logic says that the module exchange operation has to be very reliable, otherwise the purpose of on-orbit maintenance, which is to put spacecraft back into an operable condition, is defeated. Factors contributing to unreliability which are discussed include: mechanisms, latches, ability to release modules, number of degrees of freedom, number of parts, synchronization requirements, sequential effects, and the docking mechanism where it is used as part of the servicer mechanism.

To simplify the evaluation of each prospective module exchange concept, it was helpful to identify areas that will remain similar regardless of concept. For instance, module storage in the tug adapter section for a given volume of equipment will require about the same structural support for all concepts. This structural support is based on flight loads imposed by the tug vehicle. The same appears true for spacecraft support structure for the operational modules. The point is that structural weight required for the above purposes should not be charged to the particular concept. The same is true for centrally located docking devices. Of course, the addition of equipment to a docking device to afford motion was a consideration.

Mechanisms which are required to perform functions of unlatching, sliding drawers in and out, and making electrical connections such as are required in these tasks require careful detail design to ensure high reliability.

Thermal control is often brought up as being a key aspect of on-orbit servicing. There are three mission phases to be considered, each with different thermal control aspects: 1) while modules are in the stowage rack, 2) during module exchange, and 3) during spacecraft operation. When the modules are in the stowage rack they are inoperative or powered-down, thermal control coatings and insulation can be applied to the rack, electrical power for heating is vailable from the carrier vehicle, and the "barbecue" mode (slowly turning the modules with respect to the sun's direction) can be used. These techniques, which have been used successfully

on many space transportation systems such as Transtage and Apollo, are equally applicable to all servicer module stowage racks and mechanisms. During module exchange, the time should be relatively short and the module thermal capacity should be adequate to keep module temperatures within the nonoperating limits. Techniques for thermal control of the servicer mechanism need not be a screening criteria. During spacecraft operation, thermal control is the spacecraft designer's problem and is one of the reasons for not, at this time, restricting the spacecraft designer's options with regard to module sizes, shapes, and locations.

None of the servicer mechanisms evaluated appeared to require an advancement of the state of the art, so this factor is not further considered in the comparative evaluations.

5. Technical Discussion of On-Orbit Servicers

The following paragraphs present descriptions and technical discussions of the various proposed on-orbit servicer mechanisms. The technical discussions are aligned to the screening criteria which have been selected to most clearly show the comparative attributes of each servicer.

a) MDAC Direct Access Servicer Mechanism

Description: The MDAC direct access servicing concept, shown in Figure IV-4, utilizes a ring type docking device operated by eight hydraulic actuators powered by a pneumatic system. The operating sequence is shown in Figure IV-5. Pressure actuated ball latches at the end of each actuator effect the attachment of the satellite. On retraction of the actuators, the satellite is mated to the servicer, the satellite module pattern aligning with the servicer module storage pattern. The retraction motion is used to latch modules to be withdrawn from the servicer to the satellite and to latch modules to be withdrawn from the satellite to the rotating grid. A three-position rotary actuator mounted in *each* of the square segments in the rotary grid determines whether the module will be withdrawn from the satellite, the servicer or neither.

The docking mechanism actuators are then extended as shown on the second figures and the replacement and defective modules are withdrawn simultaneously. Motion at the end of the docking mechanism extension stroke automatically releases servicer modules from the satellite and latches them to the rotary grid.

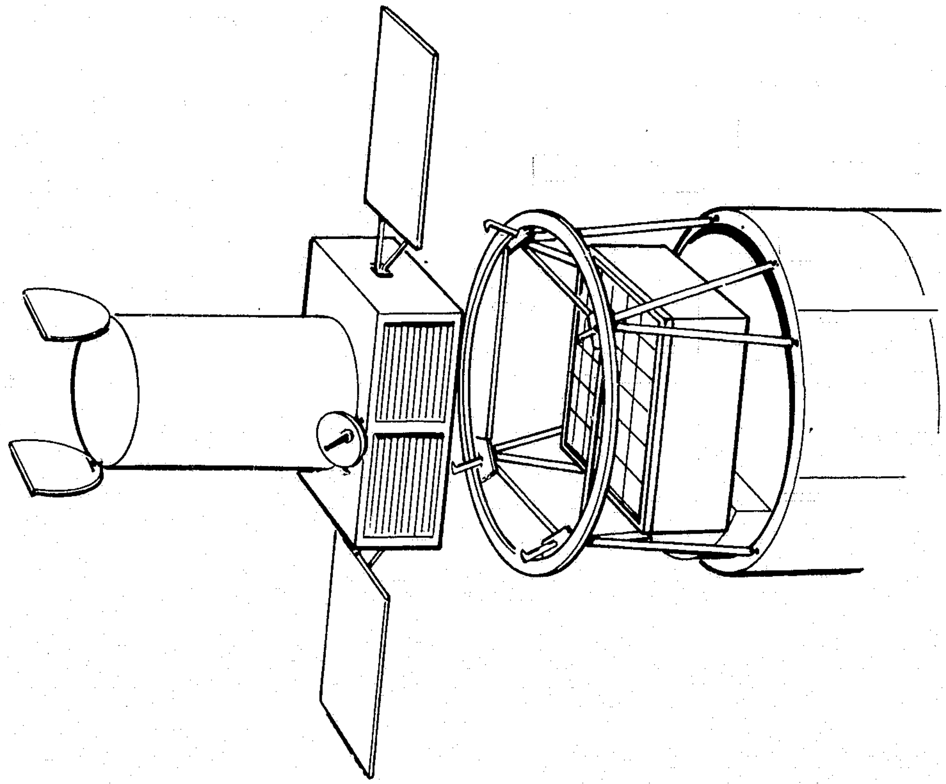


Figure IV-4 MDAC Direct Access Servicer

The grid is then rotated placing the defective modules over the storage slot vacated by the replacement module, and the replacement module over the slot vacated by the defective module.

The docking mechanism is again retracted inserting the modules in ther respective slots and locking them in place.

Half the modules can be exchanged simultaneously during one of these cycles.

Versatility: This concept is designed only to handle a given square box size which probably means that subsystems much smaller than the standard box will be packaged with other small subsystems to fill the one size box. This may be considered an undesirable compromise. Further, roll angular position accuracy is high for exchange operations. Module sizes and placements are relatively restricted.

SERVICER OPERATION

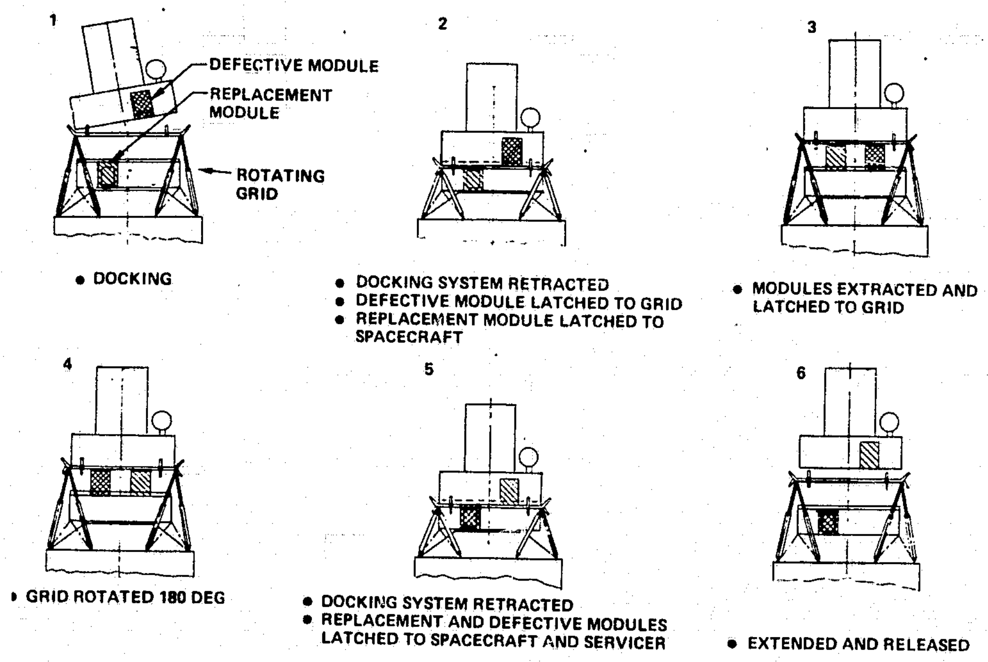


Figure IV-5 MDAC Direct Access Servicer Operation Sequence

Mechanical Advantage: This concept provides excellent push-pull capability and adequate grid rotation capability; subsequently, good mechanical advantages. Latching and unlatching requires good positive moves and available forces to have a reliable latch up. If only one or two latches are involved, it appears reasonable, but 24 latches, with 12 operating at the same time seems difficult.

Docking Mechanism: This peripheral docking mechanism is integral to module exchange and does two things that seem to be advantageous. It opens up the center for easier installation of the rotating grid table and provides a very rigid attachment to the satellite, particularly in torsion. The rotating table could still be mounted around a centrally located docking probe with some additional complexity.

For purposes of docking, this mechanism should work as well as a center docking mechanism. This is very similar to the international docking mechanism except that ball screw jacks were used in place of hydraulic cylinders. Their reasoning was that the radial arrangement opened up the center for the astronaut air-lock tunnel.

The real negative aspects of the MDAC direct access servicer appear when we start using the docking device to move modules in and out. The MDAC module exchange device depends on accurate axial travel which means that the eight hydraulic cylinders must travel forward and back in unison to maintain alignment of modules to racks. It is not apparent from the information available how this is to be done. Friction characteristics of modules moving out of the spacecraft will be different from those moving out of the rack. This may be very difficult to control. It seems that a hydraulically actuated center docking probe would accomplish the same thing much better, simpler, less weight, less cost and certainly better reliability, while maintaining good alignment.

Stiffness: This approach provides very rigid structures and good load paths are always available. For instance, the peripheral type docking device would be very stable as long as the actuation fluid system is solid. This is not entirely clear in view of the use of the accumulator since close tolerances imply need for stiff members and close position control.

Size: This concept utilizes space to the best advantage that is possible with its short (< 3 feet) stowage and operational length. The modules are efficiently positioned and are moved the least distance possible to get the modules echanged. This system adds little to overall tug length outside the module stowage depth and some to the diameter.

Weight: Since this concept utilizes the least space, it follows that it is weight competitive. The items adding excessive weight are the latching mechanisms, the rotating grid and the portions of the docking mechanism charged with axial movement. The complicated individual module latching mechanism adds up to a substantial amount of weight for this function, much more than is justified.

Reliability:

Docking Probe - The MDAC docking probe depends on proper functioning of eight actuators, eight check valves, and eight pressure-actuated ball latches plus related equipment. Each actuator requires a dynamic seal which is a high risk leak point. This large number of active parts is not conducive to high reliability.

Rotating Grid - This is a simple rotating device requiring precise radial positioning. Good reliability should be obtainable for this mechanism.

Module Latches - There are 24 modules which require latching front and back on each. There are 24, 3-way rotary selector actuators to activate the particular latch being used at that time. For instance, after dock the rotary actuator will unlatch the used module from the satellite and latch it to the rotating grid. The new module is unlatched from the tug and latched to the satellite. After dock extend, the new module is unlatched from the satellite and latched to the rotating grid. After grid rotation and dock retraction, the old module is unlatched from the grid and latched to the satellite. This adds up to four latch and unlatch operations per module. If 24 modules are exchanged, the sequence is performed 48 times for a total of 192 separate latching functions per satellite. If one latch should malfunction, a complete refurbishment of the satellite is not accomplished. Also, if a module latch failed to operate properly, the tug and satellite may be in jeopardy.

Comments: The MDAC concept is complicated, thus relatively unreliable. The concept of module exchange maintenance is based on the idea of limiting the work load to gain simplicity of equipment. This system is as complex as a six or seven-degree-of-freedom device but will accomplish only a fraction of the tasks within the capability of a general purpose manipulator.

b) Aerospace Corporation Servicer Mechanism

Description: The Aerospace servicer, depicted in Figure IV-6, utilizes a center probe and drogue docking device and hard dock is accomplished when the Apollo-type probe retracts allowing the satellite to seat on the servicer outer ring.

The servicer circular storage rack rotates 360° around the centerline enabling module locations to match any position on the satellite. There is one empty module position on the rotating rack that is first aligned with the satellite module to be replaced. The linear actuator located in the transition section extends and attaches to the satellite module, rotates to unlatch, retracts the module into the storage rack, latches the module into the storage rack, and unlatches from the module. The rack is then rotated, positioning the new module over the vacated satellite module location. The

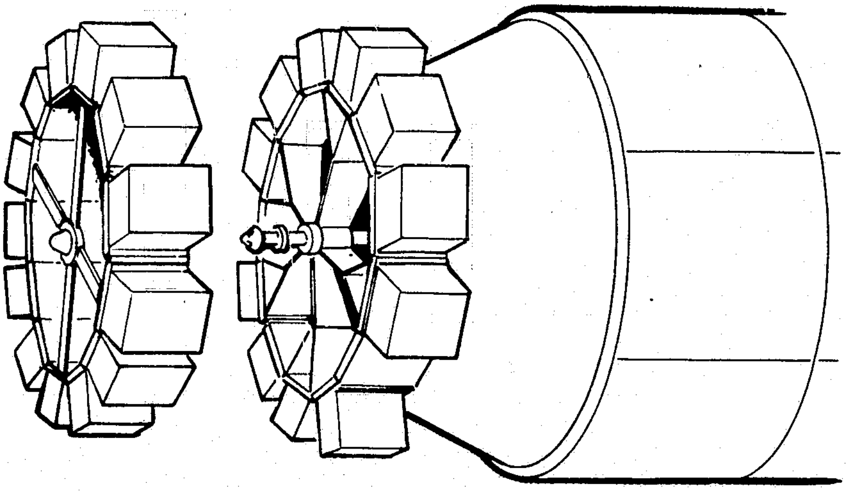


Figure IV-6 Aerospace Corporation Servicer

linear actuator for that rack extends, pushing the new module into the satellite, latching the module to the satellite, unlatches itself from the module and retracts.

This sequence is repeated for each module to be replaced.

Two versions of this concept are described in the references. One version uses a single outer tier of 16 modules, while the other adds an inner tier of 12 modules for a total of 28 modules.

This concept is also one which utilizes a minimum operating and stowage length. Modules can be located in an outer circular ring and also an inner circular ring. With modifications, the base plates can handle various width modules. Also, modules can be mounted in a circular shaped rack to improve thermal control if required.

Although in general the operation of this concept is simple, the mechanism is heavier and less versatile than other concepts. The baseplate actuators are relatively complex, and the number required (one for each servicer module) results in a heavy weight. Also the rotating frame which carries the replacement modules may adversely affect attitude control system fuel usage due to the necessity of rotating all modules for every operation. Furthermore, structural integrity may be lost between storage rack and tug adapter because

of the necessity of detachment. Extra hardware and complexity will be required to assure that the rack is secure to withstand launch and crash loads.

The requirement for circular storage patterns makes this concept less versatile than others. Also modules from an inner ring cannot be placed in an outer ring.

Versatility: This type of device does not lend itself to universal usage. Module location positions are fixed. Each satellite must match the servicing module patterns. Module sizes tend to be fixed also, however, spacecraft can be designed with one or two tiers of modules. The use of the inner tier only would suit smaller spacecraft.

Mechanical Advantage: The module indexing ring is a large diameter, presumably a large pitch ring gear driven by a small pinion. This will give a very high gear ratio for small movements and good mechanical advantage. The module cross pin mechanism being a turnbuckle type thread drive also will engage with good force; however, there is a good chance here of malfunction because of thread friction when the cross pin bottoms out. If the closing force applied is not substantially less than the opening force, jamming will occur. Good closing force is needed here to make the required electrical contact at the tapered surface. The module removal mechanism depends directly on the amount of power applied (no mechanical advantage) for positive operation. The floating module should move rather easily though, so this should work with minimum power.

Docking Mechanism: The standard Apollo docking probe is being used. This probe has been used many times and mechanically should be a sound design. It should be much simpler as the requirements are less.

Stiffness: This concept lends itself to the most efficient structural design and results in a very rigid total structure when the two vehicles are mated. There is no divergence from the circular ring frame concept. Load paths continue straight through.

Size: This concept will occupy less space than most other concepts; however, the available module volume decreases since the center area is not utilized. This may not be too inefficient if it is determined that adequate volume exists in the annular location of modules. Based on a single tier

of modules and a module depth of 24", a circular diameter limit of 168" and a box height of 24", the module volume of this concept amounts to about 150 ft^3 of volume. Assume a packaging efficiency of 13.5 lbs/ft^3, the total weight carrying capability of this concept is 1100 pounds. This could be low considering a tug payload capability of about 2500 lbs; however, with the double tier approach, the tug capability could be well utilized.

For a square center located storage rack 120" square 24" deep with center open for docking mechanism, the weight capability becomes 1340 lbs with 180 ft^3 of usable volume.

Single tug payload weight to orbit and return ranges from 2500 lbs for one spacecraft visited to 2000 lbs for three spacecraft visited. If used modules and servicer are not returned to earth, the tug can transport around 8000 lbs to geosynchronous orbit. With these higher weight capabilities, longer modules and all available end area could be utilized. The use of the full 14 ft diameter, rather than the 10 ft square area of other concepts utilizes the available shuttle cargo bay space relatively efficiently. However, the module removal mechanism occupies a volume in the transition area which has to be charged to added length of the tug to increase required cargo bay volume.

Weight: From a structural viewpoint this design is very weight efficient. The module base plate concept rather than box structure is good. However, from a mechanism viewpoint, the situation is not as attractive. Latch mechanisms are required for each module. For a 28-module arrangement, 56 cross pin latch mechanisms are needed. There is a considerable number of moving parts here of which many will probably be steel. A transport mechanism will be needed at each module station for a total of 28. There are three motions required, all of which tend to be heavy like the rack and pinion drive and rotary screw device. Motors are required to drive at least two of these and possibly a solenoid for module unlatch (56 motors and 28 solenoids, all heavy items). Therefore, overall weight appears excessive due to the actuators.

Reliability:

Docking Probe - Probably good reliability due to past flight experience. Remote docking will be a new experience but all concepts are faced with this new requirement.

Module Indexing Ring - A simple ring and pinion gear driven by a motor is used. All parts rotate on roller bearings and, mechanically, this is one of the most reliable mechanisms available. The method for determining position is not discussed but the many methods available such as potentiometers, photo cells, or limit stops would all be reliable systems.

Module Cross Pin Mechanism - This device is not a high reliability item. The index pin is close tolerance in the straight part so good electrical contact can be made. This pin can get real tight when the removal mechanism moves in to latch. Then, as the threads get loaded up, the motor which turns the screw must overcome the high friction force.

Module Removal Mechanism - The rack and pinion device for in and out movement and the push-pull unlatch device should be reliable enough. The floating rollers on each side of the base plate will allow free movement in and out. It is unfortunate that these 12 rollers are needed because that adds up to 338 rollers for all modules. It is difficult to understand their purpose as reference surfaces are provided for final alignment. If they are to provide side location, they will not be very effective as they are spring loaded. The rack must be perfectly aligned when the cross pin moves in or the electrical contact will be damaged.

General - Considering a single tier of modules and a 16 module exchange device, the total number of motors would be:

Module indexing ring	1
Cross pin actuation	16
Module removal	16
Total motors	33

Also, sixteen (16) solenoids or some type of push or pull device will be required for unlatching the removal probe for a total of 49 motors and actuators. It is assumed there will be other electrical items required such as switches to show when each function is completed prior to activation of the next function.

Comments: There is excessive electrical and mechanical equipment for the amount of work accomplished. There are eight discrete functions per module change or 16 to replace one module. For a 16 module rack, 256 discrete functions take place. Add to this 33 movements of the indexing ring

and the total becomes 289. The use of a separate mechanism to move each module forces the use of excessive numbers of mechanisms. The module removal mechanism occupies a volume in the transition area which has to be charged to added length of the tug that the oribter cargo bay must accommodate.

c) Bell Aerospace Cartesian Coordinate Servicer

Description: This servicer, shown in Figure IV-7, is equipped with a center located docking probe with total extend/retract capability. The probe

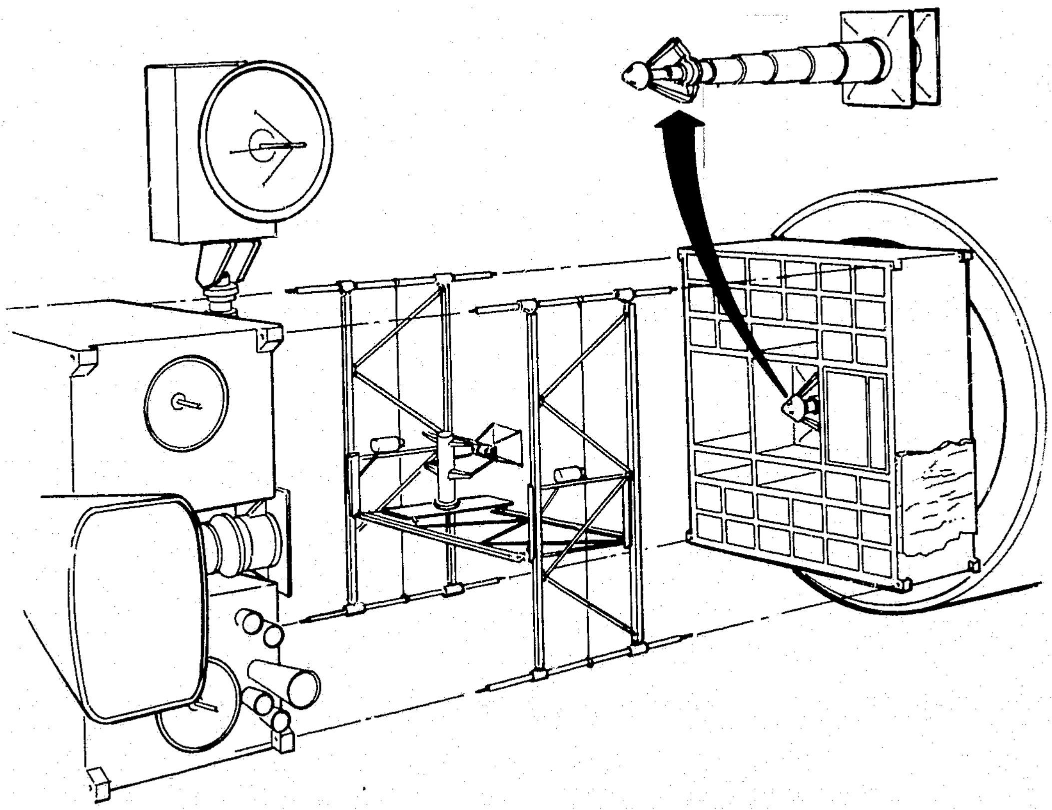


Figure IV-7 Bell Aerospace Cartesian Coordinate Servicer

extends to make initial contact, then is retracted to allow engagement and docking of the four stabilizing posts to the satellite. The docking probe is then detached from the satellite and retracted completely into the servicer to allow clearance for the satellite to operate.

The servicer mechanism provides linear travel in the X, Y and Z planes plus a rotary head to reverse position of modules. The end effector is positioned over the defective satellite module and latched through a central pin latch. The fastener wrench is rotated over one of the screw type fasteners and activated to unscrew the fastener. This is repeated for each fastener until the module is unlatched from the satellite.

The module is extracted from the satellite, rotated around to face the servicer storage rack, positioned over an empty module storage compartment and inserted. The end effector fastening mechanism then reverses operation to attach the module to the storage rack. This sequence is repeated for as many modules as need be replaced.

The docking probe is re-extended to effect undocking of the satellite from the servicer.

This concept utilizes a single end-effector for handling modules which leads to a simpler and weight-effective approach. The end-effector is also used to latch and unlatch modules. The X-Y translational carriages can locate the end-effector anywhere on the satellite end. If a rotational DOF is added near the end effector, circular storage patterns can also be used. This concept allows good versatility in various sizes and placement locations.

The major problem of this approach is the relatively complex and heavy translational carriages. Linear motion producing actuators are usually more complex and heavier than rotary actuators. Also, the operating distance between the servicer and spacecraft is large (with no easy retraction capability), and thus stowage volume in the cargo bay is large. This fact, coupled with the need for a clear center area for exchange mechanism motion, makes a long, complex, and heavy highly-retractable docking probe.

Universal Usage: This design is particularly suited to handle varying sizes and shapes of modules. This eliminates the necessity to package many small subsystems in one box to fill volume. This seems to be a requirement which should be established for the design of a module exchanger. The external docking posts, however, present a problem when docking with small diameter satellites. It is noted that a circular storage pattern is possible if an additional DOF is added. A double end-effector would provide for temporary storage making more efficient use of the rack.

Mechanical Advantage: The fact that the mechanism is strong and removes only one module at a time means that one end-effector, latch-unlatch mechanism is required. This mechanism can be provided with adequate power to handle high friction loads during module replacement. Thus a good mechanical advantage is inherent.

Screwing and unscrewing bolts by the end-effector automatically and remotely as a module attachment method seems cumbersome. It is, however, the lightest type of device. The screws add little weight and complexity to the modules, while individual latches and actuators multiplied by the number of modules adds up to a lot of moving parts. From the information available, it appears the screw installation device may be an electrically-operated impact wrench which indexes radially around the six bolt pattern circle. These are obviously low torque screws so the concept is surely feasible. It could be simpler to use one or two fasteners rather than six. The tensile strength of one 1/4" bolt is over 4000 pounds, which should be more than adequate to support a module. This would eliminate the indexing head and greatly simplify the end-effector. Also, the number of operations required to remove and replace one module is reduced from 24 to 4. Time saved and the increase in reliability is substantial. This end-effector attaches to the module with a central attachment pin. A ball-retained sleeve combined with a motor driven bolt driver was used to tension the end effector to the module.

Docking Mechanism: This docking mechanism is designed to telescope primarily to aid the attachment of the module exchanger legs to the satellite and then to retract and clear the area for operation of the exchanger mechanism. This has to be considered as a degree-of-freedom since the operation of the mechanism depends on the probe telescoping back. Telescoping devices are not simple, and they are heavy. The tube must carry wires or mechanisms to operate the probe lock, and also carry bending moments until the four legs are locked in place. Consequently, this is a very sophisticated equipment item.

Stiffness: This concept is very rigid with good load paths, providing the proper bracing is added for the docked configuration. The attendant rigidity should provide for good positioning tolerance.

Size: The square shaped modules stowed in the square rack again utilizes the usable volume most efficiently. Very little length is added to the tug and none to the diameter. However, the exchanger mechanism requires at least the same volume in front of the stowage rack, and probably some additional so that two extra module lengths are required for storage. This mechanism is non-collapsable; the orbier cargo bay must accommodate this volume. Larger volumes are required for operation; up to three module lengths.

Weight: This cartesian coordinate travel concept requires that the mechanism span the total height and width of the module stowage rack. The Y axis requires four very rigid, well-machined steel posts for the linear bearings to ride on. Each side must be driven simultaneously to prevent binding. Four heavy posts and two drives just to move in one direction is a big weight penalty. The X-axis platform reaches the full width of the rack and is wide enough to allow adequate travel in the Z-direction. All this structure must be rigid to prevent binding. The Bell drawing shows the guide posts braced along the Y-axis, and the moving table braced in the X plane. This bracing does not do much for torsional rigidity. Torsional deflection can deflect the guide posts and the moving table out of the plane of the existing braces which can cause binding in both the X and Y planes. Additional structure should be added. Too much structure is devoted to rigidizing with this concept, thus the weight will be excessive.

Reliability:

Docking - Docking of two vehicles usually consists of probe insertion and lockup. In this case, the two vehicles are brought together so the four post latches can lock up. This latching mechanism is not shown but it must consist of four separate latches. All four must work so the total docking reliability is degraded by this feature.

X-Y-Z Carriage - Providing adequate rigidity of the guide posts is possible and good synchronous drive capability of the X and Y carriages can be obtained, so the mechanism should function properly. A rack and pinion or ball screw driver is conventionally used to obtain synchronous motion. However, it appears that cables or wires are shown which is not the reliable approach.

End-Effector - There seems no reason why this end-effector could not be made reliable if the six screws per module were reduced. With six screws per module, it requires the screw removal device to operate 24 times per module for complete exchange. For 28 modules, the device must operate 672 times. If one screw failed to turn, a module would not be replaced and could diminish the effectiveness of the operation or possibly prohibit the activation of the spacecraft. If the number of screws per module is reduced from 6 to 1, the total number of operations is decreased to 112.

General: This concept should be considered a six-degree-of-freedom system: 1) docking probe, 2) X-drive, 3) Y-drive, 4) Z-drive, 5) end effector lockup, 6) end-effector indexing head and impact device. To dock, exchange 28 modules, and undock requires that 687 discrete operations take place. The docking probe alone must perform 15 operations.

This system cannot be considered very reliable in view of this large number of discrete operations. As discussed before, the number of attachment screws surely can be reduced which would help considerably. This would reduce the total discrete operations to 127 and the degrees of freedom to 5.

Comments: This concept requires too many mechanisms for the task involved. Consequently, the reliability and weight suffer. The volume required to contain the exchange mechanism is considerably more than the MDAC unit. It probably occupies no more space than the pivoting arm types during operation, but the pivoting arm types may be storable which is important when stowed in the orbiter cargo bay. The ability of the mechanism to rotate the modules, thus exposing only one side of the module to space during tansport and during use, is an important plus for this device. The thermal problem is much less under this condition. Also, electrical connections are made in only one direction.

d) RI UOP A (External) Servicer

Description: This concept, shown in Figure IV-8, utilizes a center probe/drogue docking device similar to the Apollo type. This opens up all areas to the outside of the satellite which this concept can use.

The module exchange mechanism is a boom rotating around the docking probe and extending radially past the outside diameter of the satellite and module storage rack. The boom extends and also rotates around its own centerline. The cross arm is moved linearly for a total of four degrees of freedom.

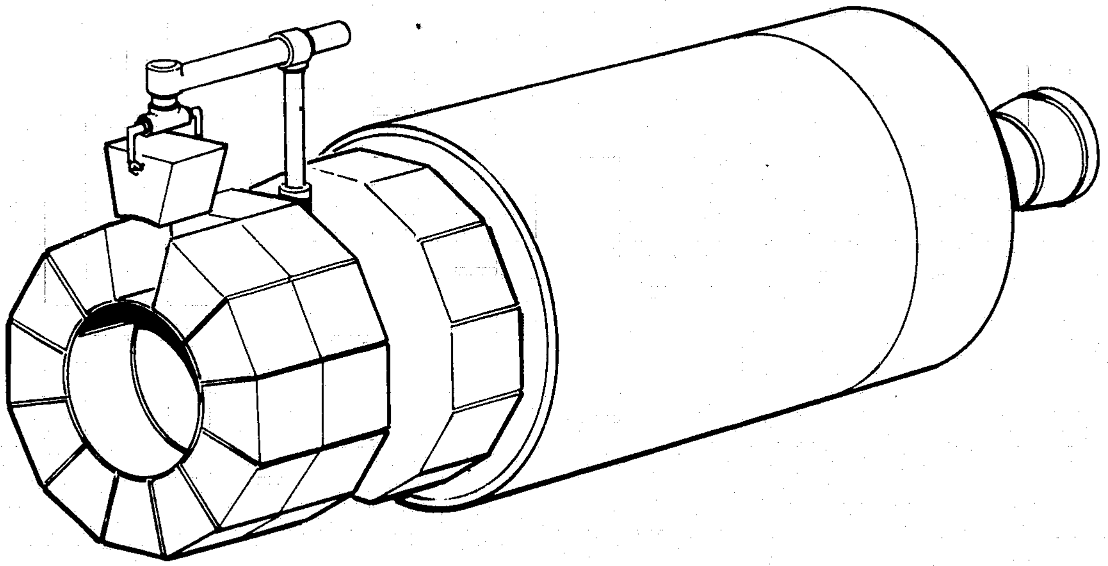


Figure IV-8 RI UOP A (External Type) Servicer

The boom is rotated radially to the angular location of the satellite module to be replaced. The cross boom is rotated and extended into position. The end-effector is moved down to the module by boom retraction and the module is unlatched and removed. The mechanism is operated to move the used module to an empty stowage rack position and latched in. The arm then moves to the replacement module stowed in the rack, removes it and installs it in the empty spacecraft module position.

This procedure is repeated for each module to be exchanged.

This concept also takes advantage of a single end-effector and begins to approach a minimum degree of freedom system (and is thus relatively simple). Since modules are exchanged outside the outer spacecraft servicer diameter, small separation and storage length is required. This concept can easily service various diameter spacecraft, and can reach one or more tiers of stowage racks.

The radial external exchange approach of this concept limits the storage pattern to an outer circular ring which is considered somewhat restrictive. With the rotating arm attached to a center pivot, a ring docking device cannot be used, and the arm weight will be greater than other pivoting arm

approaches. If a peripheral docking mechanism is used, then the radial boom must be attached to an outer ring. This results in a heavier system and one that is harder to make retractable for cargo bay stowage.

Versatility: This servicer will exhibit good acceptability for modules of various dimensions, due to the external mounting and access and no requirements to operate within the docking mechanism or internal spacecraft or carrier vehicle structural envelope. A potential restriction, particularly applicable to larger diameter spacecraft is the inability of this concept to retract modules axially from the spacecraft ends.

Mechanical Advantage: Overall mechanical advantage should be good as long as higher force motions are accomplished along the axes of the servicer actuators. Some degradation is recognized from the requirement that the arm extend across the docking mechanism and over the exterior radial surface of the spacecraft to withdraw modules radially. The comparatively long linear motion actuator may be a problem.

Docking Mechanism: Interaction with the docking mechanism for complete peripheral module access dictates that the central docking approach be used. If the docking is central, this servicer could accommodate a large degree of variation in docking mechanism designs. Since module access is not possible from the spacecraft ends with this system, a fairly large area for a docking footprint could be available.

Stiffness: Due to servicer arm segment lengths, this device will exhibit flexibility proportional to length unless corresponding structural weight is added. However, it appears that good stiffness is possible with this approach without excessive structural impact, since the spacecraft periphery is accessed from the carrier vehicle radius only. Should the drum rotating around the docking mechanism axis be larger, the radial extension of the mechanism could be even less; thus, a good stiffness property is available.

Size: This device will cause little axial length penalty in the orbiter cargo bay since it is deployed radially. The only standoff between the spacecraft and carrier vehicle would be the docking mechanism plus the diameter of the radial arm of the servicer. Some diameter extension due to peripheral stowage will be necessary, but small. The mechanism could be made retractable to within the center of the stowage module for some increase in complexity.

Weight: Weight for this servicer will be very competitive to other arm types. Total weight of the combined docking mechanism and servicer will be good since docking standoff is minimized and the servicer can be deployed from a spacecraft radius only. The inclusion of linear actuators and guideways, however, is a compromise over other servicers capable of external relatively large reach area access.

Reliability: This servicer accesses a good spacecraft surface area with little wasted motion. The requirement for a rotary table around the docking mechanism and possibly two linear actuators appears to compromise operational confidence over systems which do not require this approach. The system involves four degrees-of-freedom typically, two of which are linear. Alignment by automated means could require some sophistication in position or indexing sensors, but control of the degrees of freedom themselves appears straightforward. The combination or interaction of the rotary drum around the docking mechanism creates a related failure mode where other approaches are independent in this respect. The single mechanism to latch, unlatch and handle modules is a good reliability advantage.

Comments: This approach uses a single exchanger mechanism to handle all modules and thus should rank among the more promising concepts. The concept is more suited to a central docking mechanism, although it can be reconfigured to work with a peripheral docking system. It can also be made to be as short in stowage length as any other concept by folding and retracting the boom. The main disadvantages are the boom length and the linear actuators.

e) Pivoting Arm Type Servicers

Description: A number of servicer concepts may be classified as pivoting arm types. These include the RI UOP B (Internal), MSFC and TRW pivoting arms, and Bell Aerospace cylindrical coordinate servicers. These concepts are summarized in Figures IV-9 and IV-10.

Each of these servicers operates around a centrally located boom around which the mechanism rotates to cover the end surface of the spacecraft. They may be used with either a center probe/drogue or outer ring type docking mechanism. The radial boom can rotate and accommodate linear travel. The end-effector is at the end of a linear extender which is mounted to the boom by a rotating joint.

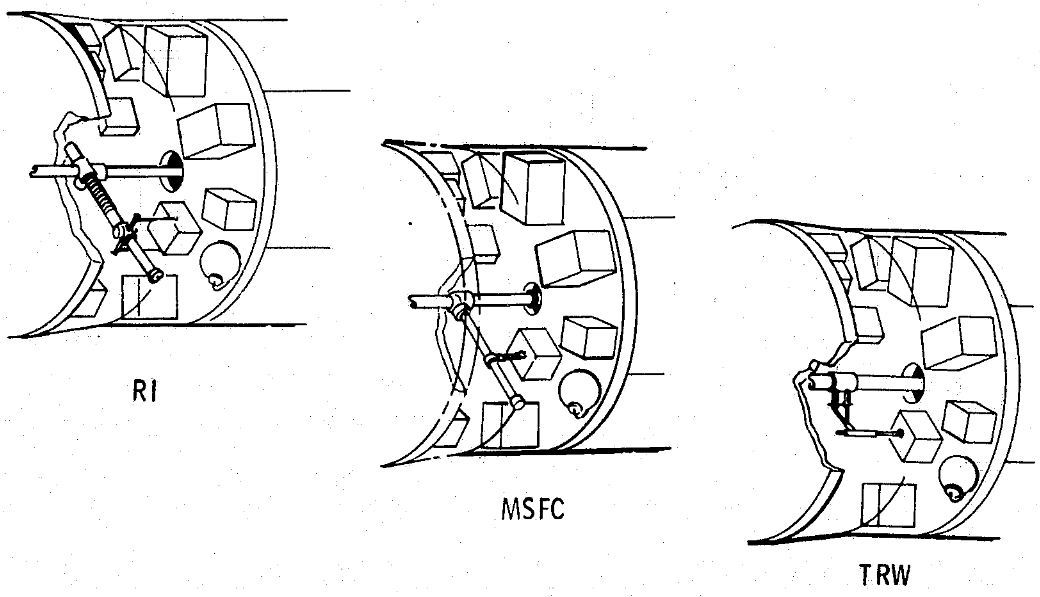


Figure IV-9 Pivoting Arm Type Servicers

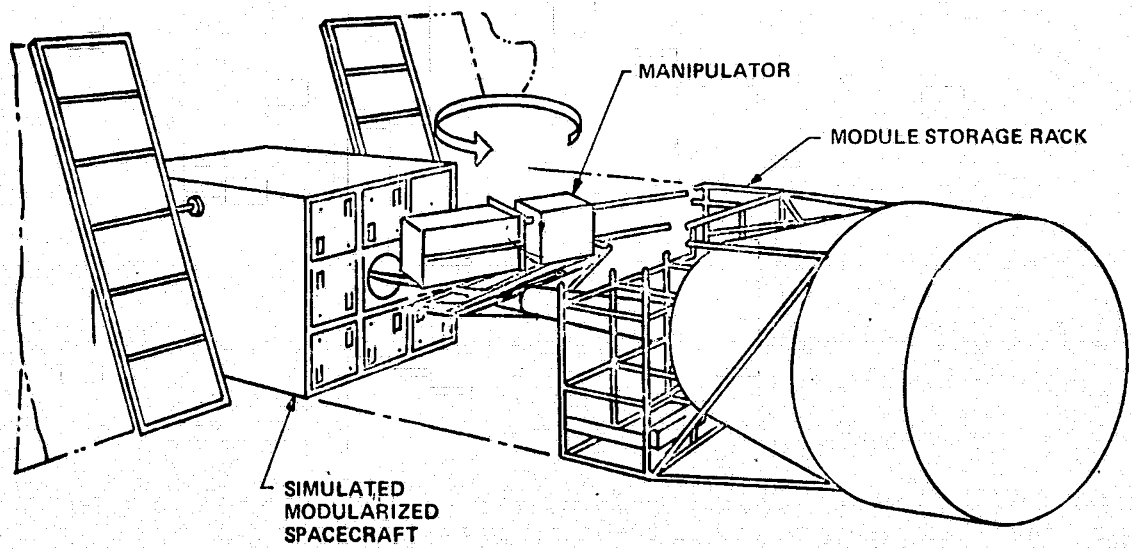


Figure IV-10 Bell Aerospace Cylindrical Coordinate Servicer

To operate, the radial boom rotates to the angular position of the used module and extends to position over the module. The end-effector rotates to align with the module latches. The end-effector extends to latch onto the module and unlatches the module from the spacecraft. The radial boom then travels away, removing the used module from the spacecraft. The radial boom can then rotate around itself, so the module is now facing the stowage rack of the servicer. The mechanism then rotates and extends or retracts to place the used module over an empty module position in the rack. The module is moved into the empty space, latched and the end-effector disengaged. The mechanism then travels to the replacement module in the stowage rack. The procedure is duplicated for installation in the spacecraft. The complete procedure is duplicated for each replaced module.

These mechanisms, which also use a single end-effector that can latch/unlatch modules, are basically minimum degree of freedom systems, yet they have good versatility in handling various sizes and locations of modules. They can be designed to stow in a short length and be reasonably lightweight.

The primary problem with these concepts is that they require relatively large separation distances between satellite and servicer during operation. This results in a long docking probe or ring docking device which may need to be retractable to save stowage space in the cargo bay. This space can be held to approximately one module length separation if a combination of rotational and translational motions are used while transferring the modules.

TRW, Rockwell International, Bell Aerospace and MSFC all have proposed a similar concept with variations. This concept is an arm extending from and rotating around the centerline of the tug and spacecraft. The arm has a radial reach to the outermost modules, has in and out and rotational motion of the end-effector for removing and positioning modules. They are all essentially the same concept and will, therefore, be evaluated together except that the Bell Aerospace cylindrical coordinate servicer is discussed separately. The TRW concept is used to represent this class because of its greater versatility, greater use of rotary as opposed to linear motions and the higher level of definition available.

Universal Usage: With linear travel along the radial boom, all points on the spacecraft end are reachable. Circumferentially, the arm can index to any place so there can be various sizes and shapes of modules positioned as desired.

Mechanical Advantage: Of the four degrees of freedom of these devices, only one, moving the module in and out, requires high forces along with the module latching mechanism. Adequate devices such as screw jacks or cylinders can be applied here to do the job with excellent mechanical advantage and attendant force transmission.

Docking Mechanism: These concepts are not dependent on the docking mechanism to aid module transfer, so either the center located or the peripheral types would be a consideration. These concepts require the largest spacing between spacecraft and servicer which means that the docking mechanism will be long compared with other methods. This is because of the volume required to maneuver the modules after removal from the servicer and prior to installation in the spacecraft.

Stiffness: These concepts are the most flexible structurally of all. The long docking mechanism and the radial boom have lengths of about 5 ft. With the added flexibility, initial close fits of guide rails, latches, etc. should be avoided. This should not be particularly restrictive.

Size: A significant feature of these mechanism is that they tend to be foldable or stowable so orbiter cargo bay volume is not wasted. Space must be allowed between the module stowage rack and spacecraft during module exchange to allow maneuvering room for the modules. This is not particularly bad as long as rigidity is maintained through the docking mechanism.

Weight: These designs are weight-efficient since the one arm accommodates all modules. It is also centrally located to minimize reach. Structure is located only where the work is being done. If the arm can be designed with a rotation motion around the center, a motion in and out at the wrist (for pulling out modules), and a wrist pivot capability (to look at the servicer and spacecraft), plus a wrist rotation to align with the modules, then the total mechanism required amounts to the four mechanized joints. The docking probe must be long enough to allow for module manipulation clearance. Unless the probe is retracted (which amounts to an additional degree of freedom), the additional length would occupy volume in the orbiter cargo bay.

Reliability: The most appealing aspect of these designs is the single mechanism required to latch, unlatch, and transfer all the modules. The number of electrical and mechanical parts required to do the job is reduced enormously. For example, the Aerospace Corporation servicer required 56 motors and 28 solenoids. Typically, this should be reduced to about 5 motors or actuators which implies a large increase in reliability.

Comments: It is apparent after reviewing all proposed concepts that a single exchanger mechanism to handle all modules is the most economical method for accomplishing the task. The methods discussed in this section and the RI UOP A (External) servicer meet this requirement

The thermal control requirement may suggest that it is desirable to reverse the modules so the mounting interface is the same in the servicer as well as the satellite. With variations, either the RI UOP A (External) servicer or the four concepts discussed here can be made to meet any of these conditions.

The Bell Aerospace cylindrical coordinate servicer is depicted in Figure IV-10. A technical discussion follows below.

Universal Usage: Some capability for universal usage, allowing random module size and limited random positioning of modules, is available. If the storage rack was changed from a square pattern to one capable of receiving various module shapes, the universal usage requirement would be better satisfied. The module exchange mechanism not having a variable radial position capability is actually the most restricting part of the design. It can only service satellites near their center, not being able to reach the outer diameter of the satellite.

Mechanical Advantage: There is nothing in this design that causes mechanical advantage problems such as long cantilevered members or precise alignment requirements. The cross travel carriage length appears to be greater than necessary. All module locations can still be reached if travel is limited to one side of center. Two linear and two rotary joints plus an end-effector are used.

Docking Mechanism: The center docking probe allows docking with all satellite diameters. It is not telescoped, resulting in a simpler design. Accurate roll axis alignment is not required for docking or the module exchange function. This style requires the least number of moving parts.

Stiffness: Rigidity of the two mated vehicles depends on the stiffness of the docking probe. If the tug is docked to one of the larger spacecraft of 20,000 pounds, the probe should be a rather large diameter, but this should be no great penalty. The mechanism is short and compact and thus can be made rigid for a low weight cost.

Size: This servicer will occupy about the same volume in front of the storage rack as the Bell cartesian coordinate type. Room is required to remove the modules and rotate them 180°. The module exchange mechanism does not stow well; therefore, orbiter cargo bay volume suffers.

Weight: This concept will result in one of the lightest of those reviewed. The large X-Y-Z frame requiring good stiffness is gone. The central docking probe remains, but the outer docking posts are gone.

Reliability: This concept could be considered to include four degrees of freedom plus end effector. The movable joints are two linear joints and two rotating joints. Radial joints are usually more reliable and lighter weight than linear joints. Providing an end-effector of simple design can be incorporated into the exchange mechanism, the concept should be capable of performing the module exchange function with the smallest number of mechanisms and discrete functions. This should indicate a high degree of reliability.

Comments: This concept provides one of the simpler methods available to exchange modules. It does not have the capability, though, of reaching a variety of radial positions which limits the number and sizes of modules it can handle. When the mechanism is rotated around the docking probe, only a square is traced by the end-effector, meaning that module attachment can only be made on this line. This severely limits its use when random module size and location is an important requirement. The cross track need only extend to one side from the center. Either form of cross track requires 270° of rotational travel about the docking axis.

f) <u>General Purpose Manipulator Type Servicers</u> - Four proposed on-orbit servicer concepts that may be categorized as general purpose manipulator types were identified. These are:

1) RI Geosynchronous Platform;
2) MDAC External;
3) MMC General Purpose;
4) GE AGOES Boom.

Each concept is described below individually, while the concepts are evaluated as a group.

General purpose manipulators simply rely on the multiple degrees of freedom to allow positioning of the end-effector in any attitude for the assigned work task. By incorporating one of the many end-effector concepts for attachment to the module, latching and unlatching, the modules are simply moved out of the spacecraft and into the servicer stowage rack. New modules are then placed into the spacecraft.

The general purpose manipulator type mechanisms have six degrees-of-freedom positioning capabilities, can reach and operate where other mechanisms cannot, and are, therefore, the most versatile of all concepts. It is believed also, assuming that the one end-effector can be used to apply connector insertion/extraction forces, that manipulators (at least relatively short ones) can be designed with competitive weights and stowage volumes (e.g., they can be folded and stowed in chordal areas). Manipulators with seven degrees-of-freedom can be designed to aid in avoiding hazards (e.g., solar panels) however, control of a seven degree-of-freedom manipulator becomes more complex.

Of course, long manipulators, designed to reach to opposite spacecraft ends for example, will not be competitive in either weight or stowage length. Also, structural or thermal bending can become problems. In general, it is believed that mechanisms with lesser degrees of freedom, and therefore lighter mechanisms, will accomplish the objectives under the established ground rules, and that a general purpose manipulator type may not be required.

RI Geosynchronous Platform Manipulator

This Rockwell International concept, Figure IV-11, is both a module exchange mechanism and a general purpose manipulator. It exchanges modules as the many other mechanisms do, but because of the more universal reach, can perform other tasks such as removal and replacement of solar panels or antennas.

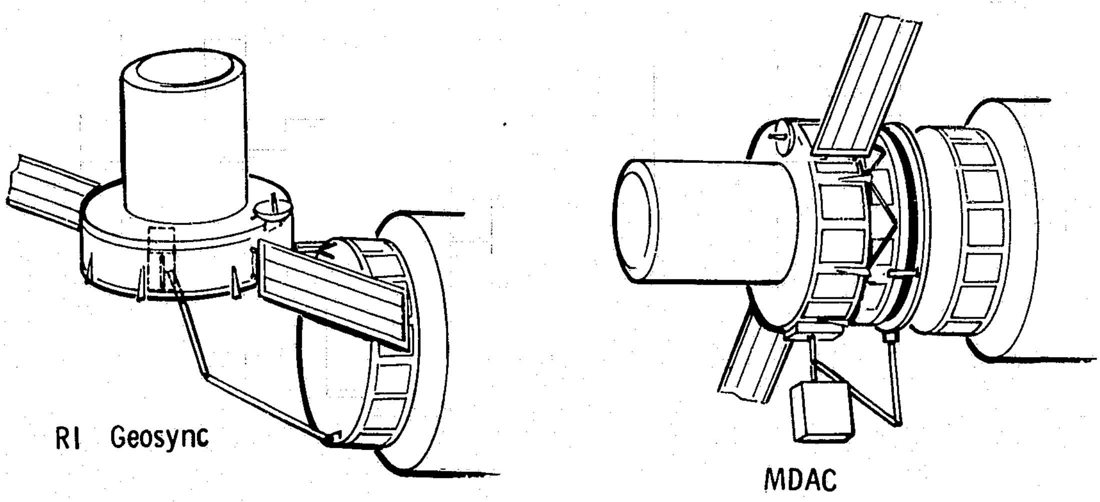


Figure IV-11 General Purpose Manipulator-RI Geosynchronous Platform and MDAC External

After effecting docking at the main ring, the ring is deployed 90° placing the spacecraft 90° to the tug servicer. This exposes the inside of the spacecraft and servicer stowage rack for the manipulator arm. A second arm may be used to position a TV camera which allows remote viewing.

Replacement operations consist of unlocking the module from the spacecraft structure, moving it inboard radially and moving it into an empty servicer stowage compartment. A new module is extracted from the stowage rack and placed in the empty spacecraft module space.

The procedure is repeated for each replaced module.

This system is intended for remote operation from the ground, not automated operation.

MDAC-External Manipulator

This servicer, also shown in Figure IV-11, can remove modules radially from the spacecraft rather than from the end. It consists of a motor driven carriage moving on a track around the periphery of the spacecraft. On the

carriage is mounted a short articulated arm capable of reaching the outside of the spacecraft and module storage rack.

The carriage travels around to the angular location of the used module to be removed. The arm positions the end-effector over the module, attaches to the module, unlocks the module and removes it from the spacecraft. The carriage then travels around to the angular location of an empty storage space on the storage rack where the module is inserted and latched in. The device then travels to the new module location in the servicer and transports the new module to the empty spacecraft space. This procedure is duplicated for each replaced module.

This concept is proposed to be used with the outer docking ring; however, the center is open allowing for the use of an Apollo-type center docking mechanism.

MMC General Purpose Manipulator

This servicer concept, shown in Figure IV-12, can remove modules axially

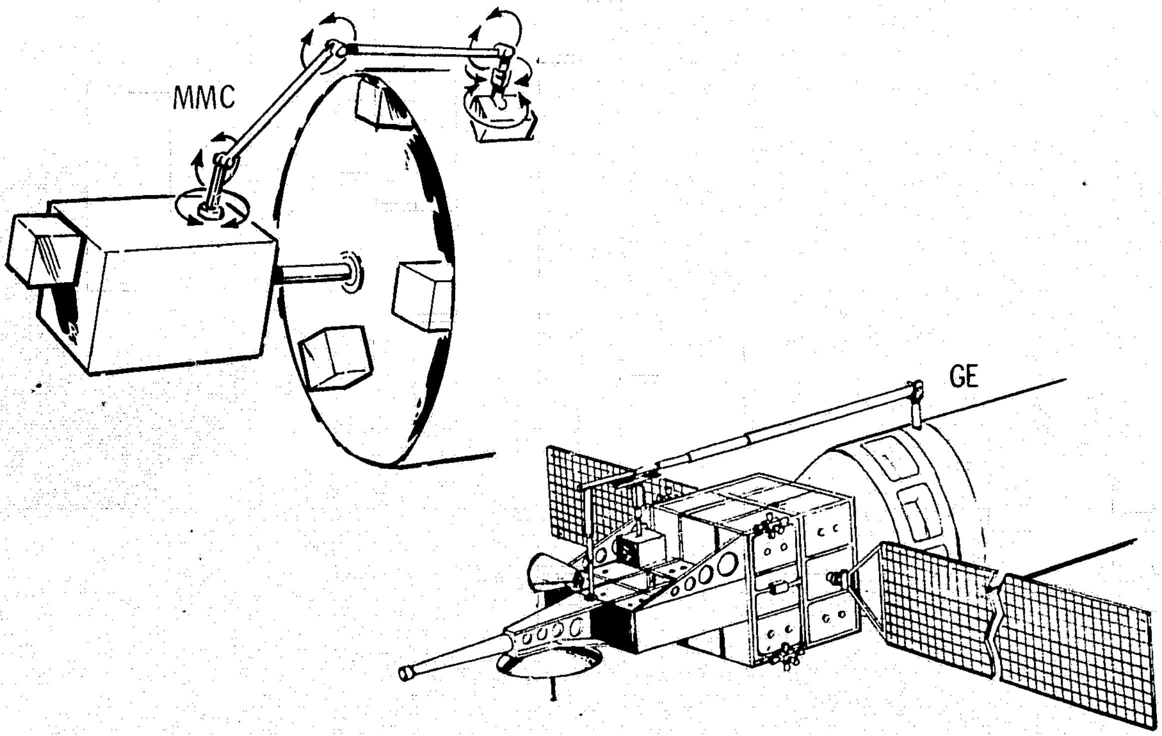


Figure IV-12 General Purpose Manipulators--MMC General Purpose and GE AGOES Boom

from the end of a spacecraft or radially from the outer surface of the spacecraft. It is shown with a full six degrees-of-freedom and mounted on a small EOTS spacecraft. The combination of small EOTS and long docking probe permits module exchange from the spacecraft end. When the carrier vehicle diameter approaches or exceeds the spacecraft diameter, then module exchange from the spacecraft end forces the use of a long docking probe. An end-effector would be fitted to the arm to hold the modules during transfer and to latch and unlatch the modules. Module stowage could be provided on the accessible exterior surfaces of the EOTS.

The module exchange sequence for this manipulator is similar to that for other manipulators. An empty position is required in the module stowage rack for temporary module stowage.

The concept is shown for a central docking mechanism but is adaptable to a peripheral docking mechanism.

GE AGOES Boom Manipulator

This servicer concept, also shown in Figure IV-12, can remove modules from four of the six spacecraft surfaces (not from the ends). This extends the two external module tiers to the full length of the spacecraft.

The long boom represents a track along which a carriage moves from the module stowage locations to the spacecraft module locations. The carriage supports an extendable member that reaches in to attach to the modules, unlatch them, and withdraw them outside the spacecraft. This extendable member also moves the modules in and out of the stowage rack. The long boom and spacecraft can be independently rotated about the tug roll axis so that four of the spacecraft faces and all of the stowage rack locations can be accessed.

The module exchange sequence for this manipulator is similar to that for other manipulators. An empty position is required in the module stowage rack for temporary module stowage. The concept is adaptable to central and peripheral docking mechanisms.

Representative of the Category

The MDAC external manipulator has been selected to represent the general purpose manipulators. The RI geosynchronous platform manipulator involved a very special docking mechanism (90° hinge) and insertion/withdrawal of modules

to the inside which seemed unduly restrictive. The MMC concept was not too well defined and would have had to be reconfigured for the tug as opposed to the EOTS so it would be comparable with the other maintenance concepts. The GE AGOES boom concept involved a traveling carriage and seemed comparatively complex for what it could accomplish.

While not shown explicitly in Figure IV-11, the MDAC external manipulator is considered to be a full six-degree-of-freedom manipulator with a capability to reach more than one tier of modules on the spacecraft surface. The detail technical evaluations are with respect to the MDAC external manipulator considered as a full six-degree-of-freedom system.

Versatility: The fact that this arm is operating outside the confinement of the spacecraft diameter, and not constrained to a particular X-Y coordinate system means that it can pick a variety of work positions around the spacecraft. If the spacecraft designer decides he wants some small modules and some large modules, or modules of different shapes, it is possible with this system. The bad feature is the inability to reach inside the spacecraft diameter for end placement of modules. The ability of the arm to reach large areas and random positions, other than the spacecraft end, gives it good universal usage capability.

Mechanical Advantage: Usually arms are equated to low tip forces and high joint torques. This is not the case here. First the arm segments are very short. Second, the angular travel of each joint will be very low compared to manipulators as used for orbiter cargo bay operations. The shoulder joint need only travel about 30° and the elbow possibly 60°. This allows the use of pneumatic, hydraulic or screw jack type actuators rather than rotational torque devices such as motors. High forces for module removal can be obtained if desired. As small deflections at the joints become large movements at the end-effectors, it is important to minimize backlash as this adds to the undesirable end-effector movements.

Docking Mechanism: This modular exchange system is adaptable to central and peripheral type docking mechanisms and might be much better off with a center docking mechanism rather than this one located right in the way of the carrier ring and arm. The length of the tug with stowage rack would be shorter. There is no requirement for accurate roll positioning of the spacecraft to tug as is required by the MDAC direct access servicer.

Stiffness: Normally manipulator arms are more flexible than the indexing rack concepts, but in this case the arm will be so short, rigidizing should be easy. Mechanism flexibility should be no problem here. Use of a peripheral docking mechanism makes for good load paths and a stiff docked configuration.

Size: This concept moves modules in and out radially rather than axially which means that room is not required between the spacecraft and servicer for mechanisms or maneuvering room. This arm is shown mounted on the outer diameter of the tug. This is the most logical place as the arm requires substantial room to maneuver. MDAC shows the arm stowed within the 14 ft diameter tug envelope so arm stowage does not seem to affect diameter or length substantially. It seems this concept could be shortened considerably from what is shown. They are using the same docking mechanism as was used on the MDAC direct access design which is a peripheral device. This forces all the docking mechanism forward of the stowage rack. If they had used the Apollo-type docking device, the two vehicles could be moved closer together as the Apollo device is buried in the middle.

In addition, the drawing shows the arm reaching outside the equipment module areas of the tug and spacecraft which for module exchange purposes does not seem necessary. This concept could be designed with a much shorter and smaller exchanger mechanism.

Weight: Exclusive of the docking mechanism and module rack structure which will not affect the comparative total weight of this concept, the weight is located in the carrier ring and arm mechanism. The carrier ring tends to be heavy because of the large diameter ring gear and arm carriage which must be rigid. The arm, if made short to reach only the modules, will be very light. The mechanism thus could be considered a medium weight device and is competitive with any of the other devices proposed. Most of the weight is allocated to the drive joints. If high tip forces are desired, the weight increases.

Reliability:

Docking Probe - The four point attachment mechanism requires eight actuators, eight check valves and eight pressure-actuated ball latches plus related equipment. All this equipment was used in the other MDAC concept for moving modules in and out of racks, which is not done here.

Arm Carriage - With one drive mechanism the arm can be positioned at any one of the modules. It is a highly reliable device and accomplishes much.

Manipulator - This design shows six driven joints exclusive of the end-effector. The fewer the joints the more reliable the manipulator. If we eliminate the need for such a long reach out of the module area, the lower arm extension can be eliminated. There also is no need for the extendable wrist motion if the shoulder and elbow/wrist pitch drives are used to keep the module aligned while removing the module. This would help simplify the arm and make it more reliable. The fact that the arm is more flexible than most concepts reduces the possibility of misalignment and possible jamming occurring during module removal.

Comments: The external manipulator servicer has many good features, the most important being that only one mechanism is required to replace modules rather than the separate mechanism per module which adds up to too much equipment. This concept should occupy the least volume in the orbiter cargo bay. There could be many variations of the arm configuration.

g) Shuttle Cargo Bay Only Servicer - Three references to this servicing mechanism approach have been found. They are:

1) MSFC shuttle module exchanger,
2) RI EOS (Rockwell International, earth observation satellite), and
3) SPAR/DSMA EOS (Spar Aerospace Products, Ltd./Dillworth, Secord, Meager and Associates Ltd., earth orbiting satellite).

As they are quite similar, they will be described together. However, as our most complete documentation is for the SPAR/DSMA version, it will be used to represent the shuttle cargo bay only servicer.

Description: Each of these orbiter-based servicing concepts (see Figure IV-13) utilizes a module exchange mechanism (MEM), a module storage rack, and a rotary docking table for the spacecraft to be serviced.

The spacecraft is docked to the rotary table. The MEM is positioned over a new replacement module in the storage rack, attached to the module, and the module is unlatched. The module is removed from the rack. The double-sided end-effector attaches to and removes the failed module from the spacecraft.

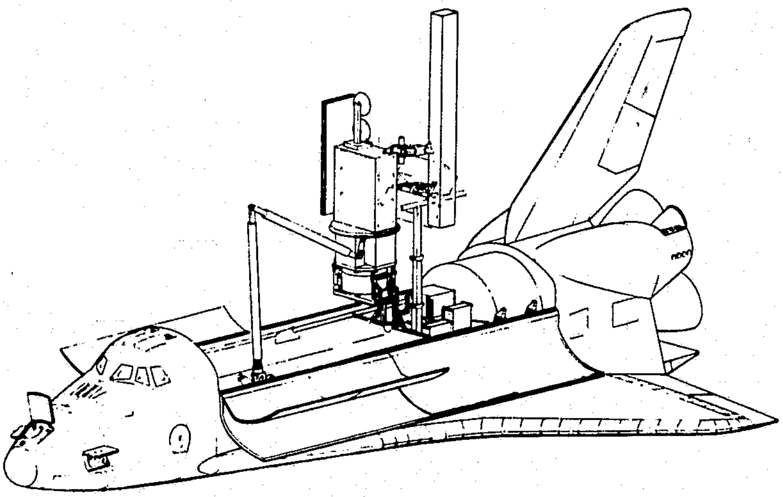


Figure IV-13 Shuttle Cargo Bay Only Servicer

The end-effector is rotated 180°, positioning the new module over the vacated module space. The new module is installed in the spacecraft. The failed module is then transported back to the storage rack and installed in the vacated module space.

The procedure is repeated as necessary for complete module exchange with the spacecraft being repositioned by the rotary docking table between each module exchange.

The servicers proposed for the use only in the cargo bay are fairly versatile in that they can service almost any size module placed almost anywhere on the spacecraft (except the spacecraft ends). An extreme penalty, however, is paid in terms of weight and stowage volume, primarily because the mechanism is long and large such that it can reach a good distance from the cargo bay.

The cargo bay servicers, of course, cannot be used in high earth orbits. It would be cost-effective to be able to use the same servicer in high earth orbits as in low earth orbits.

The GSFC approach to use the MEM includes reconfiguration of a series of spacecraft, led by the earth observation satellite, to be assembled from standardized subsystem modules and mission equipment modules that use the same latching and attaching concepts. The subsystem modules are large, on the order of 4 ft x 4 ft x 1.5 ft, and 500 lbs, which necessitated a MEM designed to handle these large modules. GSFC has had a MEM demonstration model designed and built and evaluated. The evaluation indicated the need for a number of design modifications.

Versatility: The rotating storage rack can be designed to accommodate random sizes and shapes of modules. The MEM can place modules in a variety of positions fore and aft and radially on the satellite. The limitation occurs in its inability to service the end of the satellite. Elimination of the satellite end as a module location limits the satellite design if all types of satellites are to be accommodated. Use of the SRMS for outsize modules increases the versatility of the approach.

Mechanical Advantage: The mechanical advantage of the telescoping arms depends on the type of drive mechanism involved to move them up and down. Drive screws or rack gears can provide adequate force but can become heavy. Temporary module storage is obtained by use of a double-sided end-effector. The system has been designed for easy removal and installation in the orbiter cargo bay.

The demonstration model uses chain drives, cables, rollers on as-rolled tubes, racks and pinions, electric motors, air motors, and air cylinders. The flight article design would have to be more sophisiticated. The extensive use of linear motions results in a heavy and complex mechanism. The latching system uses four separate latches for each module, one latch at each corner, a set of guides at each corner, and four attachment pip-pins between the end-effector and each module. The use of four of each element, rather than one or two, implies a high possibility of binding of the modules on insertion and withdrawal.

Docking Mechanism: In this case, the rotating table is the spacecraft docking mechanism. If it is designed to handle a variety of spacecraft diameters, which it should be, then the docking concept is adequate. Since the spacecraft end is not reachable anyway, the type of mechanism does not affect the operation of the exchanger.

Stiffness: Except for the long telescoping module exchange mechanism, the concept can be designed very rigid. The MEM telescopes up to an unsupported length of about 32 feet plus the rotating arm length of an additional 10 feet. This structure must be designed heavy to reduce deflections at the end-effector. The rigidity requirements for this concept are high because of the use of redundant latches, guides, and attachments.

Size: The rotating magazine fills the full diameter of the cargo bay and by rotating, exposes all modules for removal from the top. The exchanger occupies little of the cargo bay length and stores over the top of the magazine. The system is large because it is designed to handle large modules for large spacecraft. The magazine design appears to use only the periphery of the magazine leaving the center open. The effect is a magazine volumetric efficiency of 27 percent. When stowed in the cargo bay, the system can be mounted so that space is available to bring other spacecraft up to shuttle orbits. The overall length of the system is 15.4 feet as stowed in the orbiter cargo bay. This is somewhat larger than other concepts. The design data available to us indicates that the system is too large and heavy for stowage with a tug in the orbiter cargo bay.

The size question also involves the practicality of module exchange with the dedicated mechanism compared to module exchange utilizing the SRMS.

Weight: The spacecraft is mounted forward in the cargo bay. It is mounted on the rotary table which in turn is mounted next to the magazine. The launch loads imposed by the spacecraft must travel through the rotary table (also called the flight support system) and then into the cargo bay structure. This load path requires more structure than is desirable. The large and rigid MEM tends to be heavy. This is also reflected in the modules and latches which are quite heavy. GSFC has stated that the MEM should be strong and rigid enough to handle modules in one-g. This further adds to the anticipated weight. Their present estimate is 4743 lbs. This is heavy for tug operations and may not be acceptable for orbiter operations.

Reliability: The main point for the reliability of this system is that duplicate mechanisms are not required for the exchange of each module. The arm with its three degrees-of-freedom plus end effector accomplishes the exchange of all modules with one set of mechanisms. However, the degrees-of-freedom of the magazine (one) and of the docking table (two) must also be considered. The result is a full six degrees of freedom are involved, yet

only four controllable degrees of freedom are available for module insertion or withdrawal. Roll about an axis parallel to the orbiter X-axis, and lateral motion are not available. Structure design must be such as to accommodate tolerances for these two motions.

Use of two columns and, in later versions, two horizontal rails means that synchronization systems are required for both drives. As the deployed mechanism can inhibit closing of the orbiter cargo bay doors, an additional reliability requirement results. A backup explosive-activated separation system is to be provided.

The module attaching/latching systems each involve four mechanisms operating in parallel. If any one of these fails to open, then the system will not separate. Thus the present design implies a lower probability of orbiter survival than might need be.

Comments: The cargo bay only servicer has been conceived in a form to take full advantage of the orbiter capabilities of weight, volume, and control station location. The rotating magazine looks like a useful method for module storage and it can be made easily removable so the cargo bay can be opened up for other use at the penalty of some wasted space. However, the size and weight of this concept suffers in comparison with the smaller, lighter systems designed for tug application. The availability of the SRMS and its use for the outsize modules might be extended to reduce the size, extension, and thus weight of the module exchange mechanism.

6. Evaluations

The servicer mechanism top level evaluation is primarily concerned with: 1) identifying and understanding servicer relative merits, 2) grouping similar types of servicers, and 3) identifying which servicers are highly dedicated so they can be de-emphasized to avoid undue waste of contract resources in subsequent evaluations.

Previously the 15 on orbit servicer mechanisms were tentatively grouped as shown in Table IV-7. The analyses of the fifteen servicer mechanisms has proven the grouping to be realistic. Table IV-12 lists the 15 servicer mechanisms, and the seven servicers that have been selected as representatives of separate groups are indicated. The servicer mechanism evaluation from this point forward will concern itself with the seven groups.

Table IV-12 Servicer Mechanism Representatives

ⓧ	1)	MDAC DIRECT ACCESS
ⓧ	2)	AEROSPACE CORPORATION
ⓧ	3)	BELL AEROSPACE CARTESIAN COORDINATE
ⓧ	4)	RI UOP A (EXTERNAL)
	5)	PIVOTING ARM TYPE
		A. RI UOP B (INTERNAL)
		B. MSFC
ⓧ		C. TRW
		D. BELL AEROSPACE CYLINDRICAL COORDINATE
	6)	GENERAL PURPOSE MANIPULATOR TYPE
		A. RI GEOSYNCHRONOUS PLATFORM
ⓧ		B. MDAC EXTERNAL
		C. MMC GENERAL PURPOSE
		D. GE AGOES BOOM
	7)	SHUTTLE CARGO BAY ONLY
		A. MSFC SHUTTLE MODULE EXCHANGE
		B. RI EOS
ⓧ		C. SPAR/DSMA EOS
ⓧ		RECOMMENDED TO REPRESENT EACH DISTINCT GROUP

The factors covered in the servicer mechanism evaluation are:

1) Servicer mechanism design;
2) Maintenance of competitive servicers for both HEO and LEO applications;
3) Servicer capability to accommodate many spacecraft types (i.e., least restrictions on module characteristics);
4) Servicer interaction or dependence on the docking mechanism;
5) Servicer access to multiple spacecraft surfaces.

a) Servicer Mechanism Design - The results of the servicer design comparison are summarized in Table IV-13. The columns of the comparison chart were derived from the mechanism screening criteria which were discussed above. An additional parameter, number of mechanical functions, was added to display this important parameter.

This evaluation indicated that, when correlated to the mechanism screening criteria of this study, the pivoting arm type servicers appear to offer the greatest potential for economy through relatively low weight, good versatility and accommodation for a wide array of docking mechanism approaches, module and spacecraft sizes. The general purpose manipulators retain good versatility,

Table IV-13 Design Comparison Chart - Spacecraft Servicers

	VERSATILITY	MECHANICAL ADVANTAGE	NUMBER OF MECHANICAL FUNCTIONS	DOCKING MECHANISM DEPENDENCY	STIFFNESS	SIZE	WEIGHT	RELIABILITY
MDAC DIRECT ACCESS SERVICER	POOR	MEDIUM	HIGH	POOR	LOW	SMALL	MEDIUM	POOR
AEROSPACE CORPORATION	POOR	MEDIUM	HIGH	GOOD	LOW	MEDIUM	MEDIUM	POOR
BELL AEROSPACE CARTESIAN COORDINATE	FAIR	MEDIUM	MEDIUM	FAIR	LOW	LARGE	HIGH	FAIR
RI UOP A EXTERNAL	FAIR	MEDIUM	LOW	GOOD	HIGH	MEDIUM	MEDIUM	GOOD
TRW PIVOTING ARM	GOOD	HIGH	LOW	GOOD	HIGH	MEDIUM	LOW	GOOD
MDAC EXTERNAL MANIPULATOR	FAIR	MEDIUM	MEDIUM	POOR	MEDIUM	SMALL	MEDIUM	FAIR
SHUTTLE CARGO BAY ONLY SERVICER	FAIR	MEDIUM	MEDIUM	GOOD	MEDIUM	LARGE	LARGE	FAIR

size, and reliability at a greater weight and structural flexibility and lower mechanical advantage than pivoting arm type servicers. The remaining servicer types tend to be more dedicated to particular spacecraft applications; therefore, they do not offer a competitive potential for economy in application across the mission model.

In addition, more detailed comments in the following paragraphs provide further insight into the relative merits of the various proposed on-orbit servicers.

Ideally, a servicer should be capable of handling selectable-size modules in selectable locations on the end of the satellite, on the outer ring, and along the outside of the spacecraft for radial removal. This suggests a device capable of three translational motions and orientation motions resulting in 5 or 6 degrees of freedom total. The servicer should be capable of docking with small diameter spacecraft as well as the large 15 ft diameter spacecraft and performing module exchange.

A concept which provided limited usage for now as well as in the future and was at least as complex as one with more flexible capability was deemphasized for subsequent analysis. The MDAC direct access servicer and the Aerospace Corporation carousel types definitely fall into this category. These concepts require individual active mechanisms for each module exchange position. The sheer number of powered mechanisms precludes the possibility of trouble-free operation, not to mention cost and weight penalties. On the other hand, the Bell Aerospace cartesian coordinate servicer utilizes only one set of actuation mechanisms to affect module exchange. It can reach any point within the limits of x-y drive systems, meaning that within this area, modules could be of selectable sizes and shapes. This servicer, however, presents problems with docking with small diameter satellites. Both the MDAC direct access and Aerospace Corporation carousel types suffer from this limitation.

The RI UOP A (External) servicer is comparable to the Bell Aerospace cartesian coordinate design in mechanism complexity and weight. It is designed to remove modules radially at the outer ring only, but has the ability to reach any point around the circumference within reach of arms' length. This allows the spacecraft designer to select module shapes and sizes. It could also be used to maintain miscellaneous equipment which may be identified in the future, such as earth-pointing experiments. The spacecraft end, however, is not available to this servicer, which seems a serious disadvantage in view of the more immediate application to the communication spacecraft requirement for end servicing.

Several servicers rotate around the center docking probe, fold or telescope out for radial positioning and have various similar degrees-of-freedom to position the end-effector and move the modules. These include: RI UOP A (External), and the TRW pivoting-arm type. This general type of servicer uses only one set of mechanisms to handle all modules. They can be designed to service selected shaped and positioned modules. They require the least amount of structure to reach all positions because they originate at the vehicle centerline. There is only one structural member to reach the work area, no duplication. They can service small as well as large diameter spacecraft. Because of the wide range of positioning capability, the spacecraft designer has a wide latitude for module position, particularly in regard to thermal control and instrument location. The RI UOP A (External) concept replaces

modules radially from the spacecraft periphery while the TRW pivoting arm type replaces modules axially from the spacecraft end.

If any of the pivoting arm servicers, which are designed to replace modules axially from the spacecraft end, were extended just slightly to reach outside the largest spacecraft diameter, an additional or interchangeable outer mechanism could be included as the growth factor designed to meet future spacecraft design requirements. For example, the TRW pivoting arm with one rotary joint in the middle can reach from near vehicle center to a radius of 8 or 9 feet with total area coverage for only 3 degrees of freedom. A rotating head and linear movement at the end-effector satisfies all axial module removal requirements from the smallest to largest spacecraft planned. Modification of the device at the end-effector could add radial module removal capability at a later date if desired.

It could be argued that if the pivoting arm servicer described above is expanded to include five or six degrees of freedom to meet all future requirements, this approaches the complexity of a general purpose manipulator. It probably does. A general purpose manipulator mounted at a fixed outer diameter location is inefficient as it is too close to the modules mounted on that side and must reach at least 14 feet to service the far side. A general purpose manipulator mounted on a peripheral ring to provide good module access will have a higher weight. A centrally located arm need only reach the radius of operation.

Servicer interaction with the docking mechanism bears heavily on the functional versatility of the system. The two general types, peripheral devices and the center probe/drogue types, are most often proposed. The peripheral docking device normally is shown at about the 14 ft diameter location. This implies that all spacecraft, large and small, must interface at this large diameter. This is a potential penalty for small diameter spacecraft. Also, the docking ring partially blocks the outer servicing areas from the inner servicing areas.

Alternately, a centrally located docking mechanism would open up the total area for servicing regardless of the type servicer chosen. Apollo program docking studies showed the probe/drogue concept superior in many respects -- weight, dynamics, simplicity -- to other candidates. However, these concepts must be viewed in this study in light of interaction with the servicer mechanism in ways that imply various relative economic penalties or benefits.

The centrally located pivoting arm-type servicers affords the best access to spacecraft of various diameters with reasonable complexity and growth potential to meet future spacecraft servicing requirements.

b) Maintenance of Competitive Servicers for Both HEO and LEO Applications - Included in Table IV-14 is a correlation of servicer applications to orbital application as shown in the literature. Servicers one through six have been

Table IV-14 Servicer Application in Orbit

SERVICER	APPLICATION LEO	HEO	COMMENTS
1) MDAC DIRECT ACCESS	X	ⓧ	IF TUG EMPLOYED, OTHERWISE MUST MODIFY FOR CARGO BAY USE
2) AEROSPACE CORPORATION	X	ⓧ	
3) BELL AEROSPACE CARTESIAN CO-ORDINATE	X	ⓧ	
4) RI UOP A (EXTERNAL)	X	ⓧ	
5) TRW PIVOTING ARM	X	ⓧ	
6) MDAC EXTERNAL (GENERAL PURPOSE MANIPULATOR)	X	ⓧ	
7) SPAR/DSMA CARGO BAY ONLY	ⓧ	X	HAS BEEN STUDIED FOR TUG ADAPTATION

ⓧ PRIMARY MODE SHOWN IN LITERATURE

X ADAPTABLE TO THIS MODE

basically designed for tug delivered servicing missions. However, they could be modified to be utilized at the orbiter cargo bay. Some differences in modifications would exist. The SPAR/DSMA cargo bay only servicer currently has been designed for servicing of EOS in low earth orbit. However, an adaption for high earth orbit has been studied. Considerable changes have to be made to reduce the weight.

c) Servicer Capability to Accommodate Many Spacecraft Types - This factor is important when considering the projection of a servicer to the entire maintenance mission model. If this aspect were interpreted as a measure of restrictions on module metrics, then Table IV-15 summarizes the module characteristics accepted by the proposed servicers.

Table IV-15 On-Orbit Servicer Module Accommodation

SERVICER	MODULE CHARACTERISTICS ACCEPTED*
1) MDAC DIRECT ACCESS	HIGHLY STANDARDIZED, RECTANGULAR SURFACES
2) AEROSPACE CORPORATION	HIGHLY STANDARDIZED, TRAPEZOIDAL AND RECTANGULAR SURFACES
3) BELL AEROSPACE CARTESIAN CO-ORDINATE	VARIETY OF DIMENSIONS
4) RI UOP A (EXTERNAL)	REGULAR DIMENSIONS
5) TRW PIVOTING ARM	VARIETY OF DIMENSIONS
6) MDAC EXTERNAL (GENERAL PURPOSE MANIPULATOR)	VARIETY OF DIMENSIONS
7) SPAR/DSMA CARGO BAY ONLY	VARIETY OF DIMENSIONS
*REPORTED IN LITERATURE SOURCES	

The analysis indicated that servicers 1) and 2) should be deemphasized since the impact of highly standardized modules on spacecraft programs would be excessively costly. Item 4) should also be deemphasized based on its potential to require spacecraft modules of regular dimension. Since the ability of an on-orbit servicer system to accommodate modules of a variety of dimensions appears to lead to both lower spacecraft impact and a potential for multiple functions (such as contamination shield removal/emplacement, or obstacle/appendage avoidance), it is recommended that servicers selected for detailed economic comparison, based on module accommodation, include: 3), and 5) through 7).

d) Servicer Interaction or Dependence on the Docking Mechanism - Table IV-16 summarizes the interaction of on-orbit servicers with the docking mechanism. This analysis showed that most servicers could be adapted to both central and peripheral docking and appeared to provide potential for various spacecraft accommodations, at least in concept. However, from the standpoint of imposed limitations to spacecraft configuration for servicing, the central docking devices imposed less restriction than peripheral unless a good degree of dexterity is available in the servicer to avoid or work around the docking mechanism. That is, in view of carrying servicers further which seem to provide the least restrictions on spacecraft configurations, servicers 5 and 6 again appear to offer good flexibility. It is also noted

Table IV-16 On-Orbit Servicer Interaction with Docking Mechanism

SERVICER	DOCKING MECHANISM TYPE		COMMENTS
	CENTRAL	PERIPHERAL	
1. MDAC DIRECT ACCESS	X	ⓧ	DOCKING MECHANISM USED TO EFFECT MODULE EXCHANGE. CENTRAL SYSTEM WOULD REQUIRE REDESIGN OF APOLLO DOCKING PROBE TO MAKE IT CONTROLLABLE IN EXTENSION AND RETRACTION.
2. AEROSPACE CORPORATION	ⓧ	X	APOLLO-TYPE DOCKING DEVICE
3. BELL AEROSPACE CARTESIAN COORDINATE	ⓧ	X	APOLLO-TYPE DOCKING DEVICE
4. RI UOP A (EXTERNAL)	ⓧ	X	APOLLO-TYPE DEVICE
5. TRW PIVOTING ARM	ⓧ	X	APOLLO-TYPE DEVICE SHOWN; CAN ALSO WORK INSIDE PERIPHERAL DEVICE
6. MDAC EXTERNAL	X	ⓧ	PERIPHERAL DEVICE SHOWN CAUSES MANIPULATOR TO BE INDEXED AROUND ON TRACK
7. SPAR/DSMA CARGO BAY	N/A	ⓧ	USES DOCKING ADAPTER OR SWING TABLE OF CARGO BAY WHICH BLOCKS ACCESS TO END OF SPACECRAFT
ⓧ PRIMARILY SHOWN IN LITERATURE; X - CAN BE ADAPTED TO THIS FORM			

that visiting system 4 was discussed above as an extension of the TRW pivoting arm allowing access of modules through spacecraft surfaces other than the docking plane. Therefore, little loss of generality should result if the RI UOP A External were deemphasized at this point. Servicer 7 requires an end type docking adaptor which restricts access to the end of the spacecraft.

Servicers 5 and 6 are a pair of servicers incorporating good span of versatility with flexibility to accept various spacecraft designs from docking mechanism considerations.

e) <u>Servicer Access to Multiple Spacecraft Surfaces</u> - Access to multiple spacecraft surfaces without redocking is an additional measure of servicer restriction on spacecraft configuration. Accordingly, access to both the

docking plane surface as well as surfaces other than the docking plane should lead to eventual economy. Figure IV-14 shows, by the shaded area, the portions of a spacecraft external surface that the various on-orbit servicers can reach. Table IV-17 summarizes this factor for each of the servicers being evaluated. This evaluation shows that servicers 5 through 7 provide better potential for economy through less restriction on spacecraft configuration.

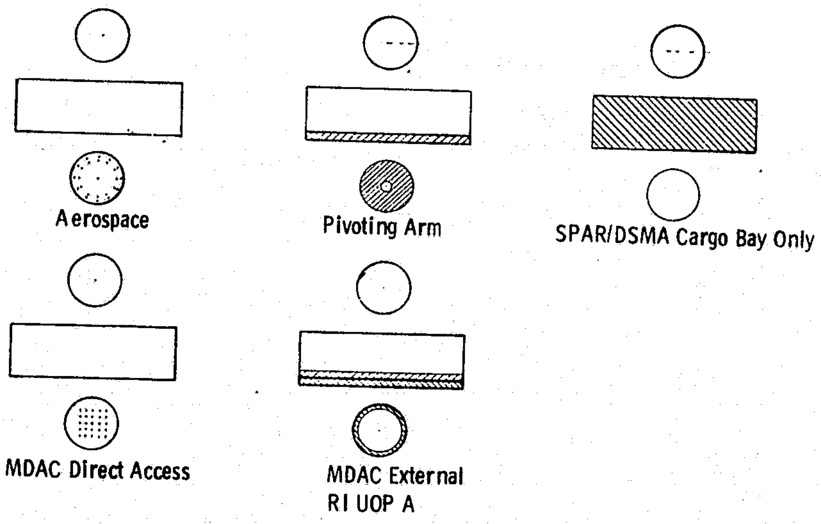


Figure IV-14 Comparative Access to Spacecraft Surfaces

Table IV-17 Comparative Access to Spacecraft Surface for Servicers

SERVICER	ACCESS TO SPACECRAFT SURFACES: DOCKING PLANE ONLY	SURFACES OTHER THAN DOCKING PLANE	BOTH
1. MDAC DIRECT ACCESS	ⓧ		
2. AEROSPACE CORPORATION	ⓧ		
3. BELL AEROSPACE CARTESIAN COORDINATE	ⓧ		
4. RI UOP A (EXTERNAL)		ⓧ	WITH ADDITIONAL DOF OR CONFIGURATION MODIFICATION
5. TRW PIVOTING ARM	ⓧ		WITH ADDITIONAL DOF AND LENGTH
6. MDAC EXTERNAL		ⓧ	IF CENTRAL DOCKING EMPLOYED
7. SPAR/DSMA CARGO BAY		ⓧ	IF SRMS USED TO REDOCK AND ORIENT

7. On-Orbit Servicer Selection

The 15 on-orbit servicers were described above and in that description it was noted that the 15 proposed on-orbit servicer mechanisms could be represented very effectively in seven categories (See Table IV-12). Of these categories, three include several concepts. Due to the close similarity of the servicers within these categories, it was determined that a single servicer concept be selected to represent these categories. The representative servicers selected should be comparatively well documented to efficiently expedite their economic comparison to other concepts. Accordingly, the following selections were made:

a) Pivoting arm-type servicers to be represented by the TRW concept;
b) General Purpose Manipulator-type to be represented by the MDAC external concept;
c) Shuttle Cargo Bay Only servicers to be represented by the SPAR/DSMA EOS concept.

In addition, the Bell Aerospace cylindrical coordinate concept appears to be an improved version of the Bell Aerospace cartesian coordinate servicer and incorporates advanced thinking regarding access to the entire spacecraft module surface, interaction with the docking mechanism, and module emplacement mechanism simplicity which leads to a more economical device. Therefore, it was determined that the Bell Aerospace cartesian coordinate servicer need not be selected for further servicer evaluations. This suggestion does not reject such a device for certain dedicated applications, but merely says that contract resources should not be expended in subsequent tasks to further compete this servicer since it can be shown to be relatively inefficient when viewed in light of application to the entire characteristic set of spacecraft at this phase of study investigations.

It was further noted that the Bell Aerospace cylindrical coordinate concept is essentially a pivoting arm-type of servicer and can be well represented for the detailed evaluations by the TRW concept. Thus, it is not deemphasized, but is represented by similarity to the TRW concept for continued economic comparisons.

The shuttle cargo bay only servicer mechanisms can be well represented by the SPAR/DSMA EOS concept for subsequent spacecraft servicer evaluations, particularly as it is the concept being pursued by the sponsor, Goddard Space Flight Center.

General purpose manipulator types can be effectively represented by the MDAC external concept. Thus, by similarity, the MDAC external servicer will represent the RI geosynchronous platform, MMC general purpose, and GE AGOES boom servicer concepts for subsequent comparisons.

At this level of the selection process, the following six servicers remain for further consideration:

1) MDAC direct access,
2) Aerospace Corporation,
3) RI UOP A (external),
4) TRW - pivoting arm type,
5) MDAC External - general purpose manipulator, and
6) SPAR/DSMA EOS - shuttle cargo bay only

The above evaluations considered these six concepts with respect to:

1) Application in orbit (LEO or HEO),
2) Spacecraft module accommodation,
3) Docking mechanism interaction, and
4) Access to spacecraft surfaces.

The shuttle cargo bay only servicer is limited to LEO applications and thus cannot be considered for HEO applications. The direct access, Aerospace Corporation, and RI UOP A (external) are all restricted to regular size modules and effectively restrict the spacecraft designer more than the pivoting arm or general purpose manipulator. They are also greatly restrictive in access to spacecraft surfaces as well as being somewhat more complex. The pivoting arm and the general purpose manipulator were thus selected to represent the whole class of on-orbit servicers for the high earth orbit case. They span the ranges of complexity - four degrees of freedom versus six; spacecraft access - axial vs radial module removal; versatility - limited to module exchange vs general purpose; and prime docking mechanism - central vs peripheral.

The cargo bay only servicer was also considered as a candidate for LEO operations only.

The advantages and disadvantages listed in Table IV-18 are based on the SPAR/DSMA data available to us and may not be applicable as the design is improved and updated. Servicing with a cargo bay only servicer appears

Table IV-18 Cargo Bay Only Servicer - Advantages and Disadvantages

ADVANTAGES	DISADVANTAGES
ENGINEERING MODEL HAS BEEN BUILT	LIMITED TO SERVICING SPACECRAFT IN LOW EARTH ORBIT
SERVICING CAN BE PERFORMED ON MULTIPLE SURFACES OF THE SPACECRAFT	WEIGHT - 4,743 lb
TOTAL MODULE EXCHANGE TIME IS LOW	SIZE - 15.4 ft LONG, 2,724 ft^3 OF CARGO BAY
MAN IS AVAILABLE FOR:	LIMITED MODULE SIZES AND SHAPES
CONTINUOUS OPERATION OF EXCHANGER MONITORING OF SERVICING STATUS EVALUATING AND HANDLING CONTINGENCIES	LIMITED MODULE LOCATIONS RESTRICTS OPERATIONS OF ORBITER DURING SERVICING ACTIVITIES
SRMS IS AVAILABLE FOR OUTSIZE MODULE EXCHANGE	HIGH MODULE GUIDE FORCES
	MODULE GUIDE FORCES REACT THROUGH ENTIRE STRUCTURE
	LIMITED TO CERTAIN SIZE & SHAPE OF SPACECRAFT
	CANNOT COMPENSATE FOR MODULE ROLL ERRORS
	MAGAZINE VOLUME EFFICIENCY = 27%

functionally to be a technically acceptable approach. A number of significant advantages are listed. A major disadvantage is its application only to LEO spacecraft. Many of the other disadvantages and advantages relate only to this specific expression of the design of a cargo bay only servicer.

When the TRW pivoting arm servicer or the MDAC external-general purpose manipulator are considered as cargo bay servicers, they appear to be more technically feasible than this particular expression of a cargo bay only servicer. Their comparable metrics are: 1130 lbs, 5 ft long, and 884 cu ft. They place fewer limits on spacecraft module sizes, shapes, locations, and masses as well as having better design details. They are also applicable to both HEO and LEO operations and thus represent potentially lower life cycle costs than if individual systems are designed for LEO and HEO. Therefore, it was determined that the TRW pivoting arm and the MDAC external-general purpose manipulator should be further investigated for orbiter cargo bay module exchange.

C. COMPARISON OF PIVOTING ARM AND GENERAL PURPOSE MANIPULATOR ON-ORBIT SERVICER

The results of a more detailed analyses of the Pivoting Arm (TRW) and General Purpose Manipulator (MDAC) on-orbit servicers are discussed in this section. It was concluded in the previous section that these two servicers most completely satisfied the first level evaluation against the screening criteria. Preliminary versions of the servicers are illustrated in Figures IV-15 and 16. The illustrations show the basic features of the concepts. A summary of the comparison factors is given in Table IV-19, and a discussion of each factor follows.

1. Module Removal Direction

The pivoting arm (Figure IV-15) uses axial removal of the modules whereas the general purpose manipulator uses radial. For each system the servicer mechanism design dictates that the module removal direction for the stowage rack be same as for the spacecraft. Module removal direction interrelates with other comparison factors like volumetric efficiency and module shapes which will be discussed later. In the geometric layouts (Chapter V) made for the spacecraft in the characteristic set it was found that all of the spacecraft could be configured to accommodate axial removal.

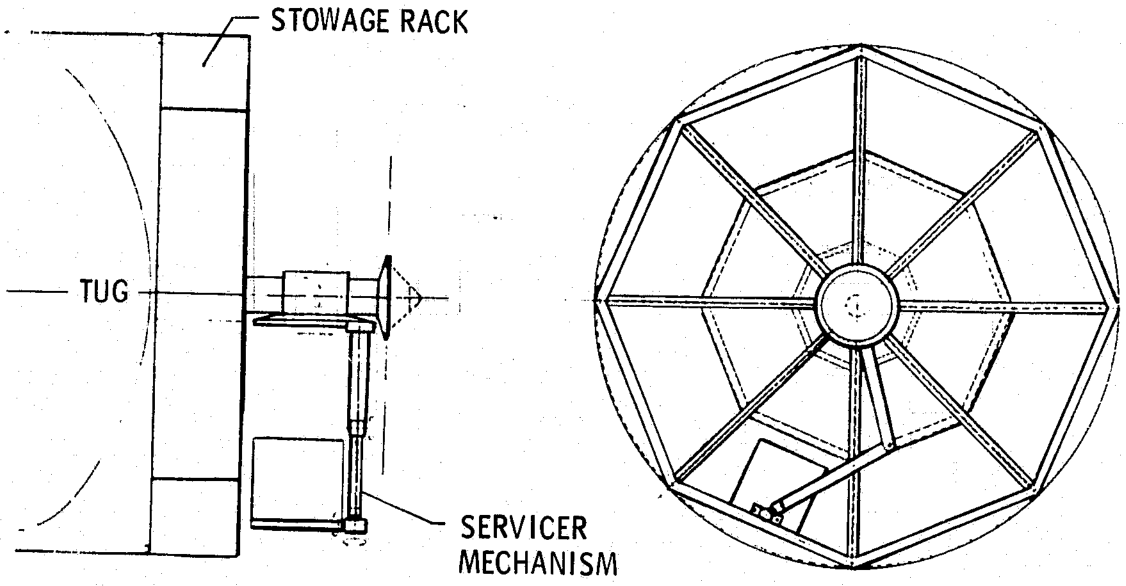


Figure IV-15 Pivoting Arm On-Orbit Servicer (TRW)

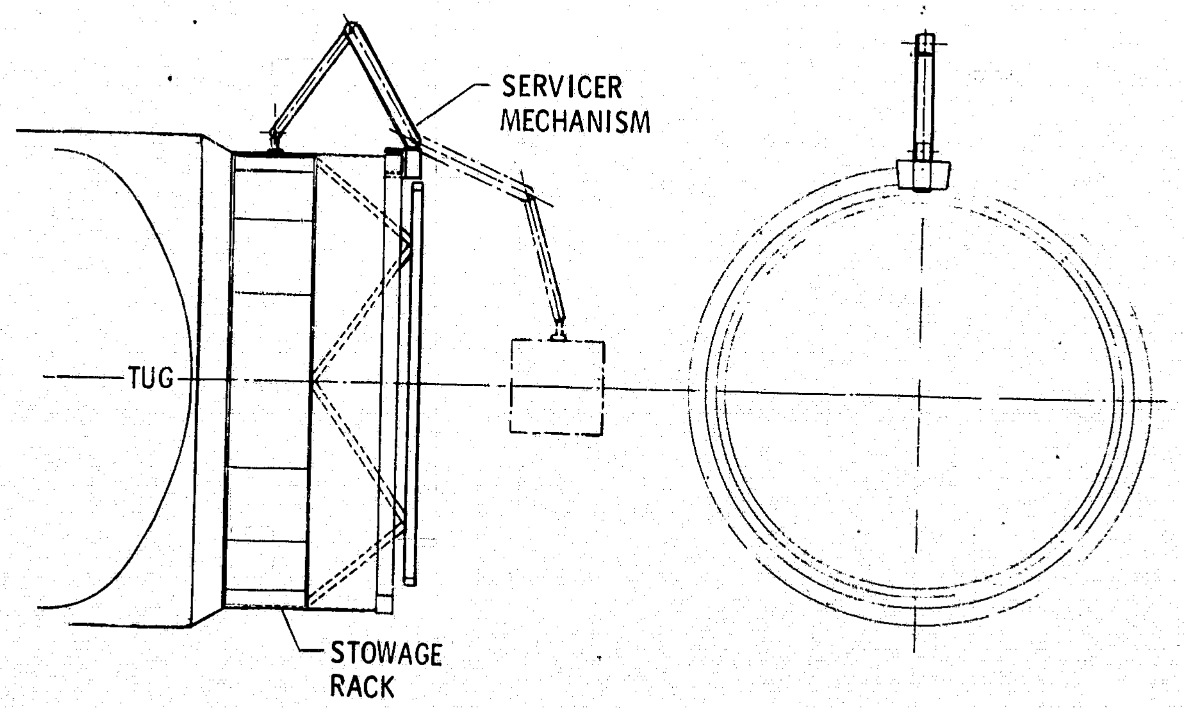


Figure IV-16 General Purpose Manipulator On-Orbit Servicer (MDAC)

Table IV-19 Comparison of Pivoting Arm and General Purpose Manipulator

COMPARISON FACTOR	PIVOTING ARM (TRW)		GENERAL PURPOSE MANIPULATOR (MDAC)	
1. MODULE REMOVAL DIRECTION	AXIAL		RADIAL	
2. SPACECRAFT LOCATION AND DISTRIBUTION OF MODULES	ANY PATTERN ON END OF SPACECRAFT: GOOD ACCESS TO CENTRAL REGION AND MORE EFFECTIVE GROUPING OF SUBSYSTEMS FOR THERMAL CONTROL		ANY PATTERN ON PERIPHERY OF SPACECRAFT: POOR ACCESS TO CENTRAL REGION	
3. MODULE SIZES AND SHAPES	GOOD FLEXIBILITY		RESTRICTED TO PIE-SHAPES IF CENTRAL REGION IS USED.	
4. VOLUMETRIC EFFICIENCY	GOOD EFFECTIVE ACCESS TO CENTRAL REGION PERMITS GOOD PACKAGING		POOR - INEFFECTIVE ACCESS TO CENTRAL REGION	
5. MECHANIZATION COMPLEXITY - DEGREES OF FREEDOM	SIMPLE FOUR DEGREES: ROTARY - 3 LINEAR - 1		RELATIVELY COMPLEX SIX DEGREES: ROTARY - 5 LINEAR - 1	
6. SERVICER WEIGHT (lb)	TUG	ORBITER	TUG	ORBITER
SERVICER MECHANISM	100	100	250	250
STOWAGE RACK	400	400	400	400
TUG ADAPTER	50	N/A	80	N/A
SUPPORT EQUIPMENT	N/A	450	N/A	450
CONTROL ELECTRONICS	30	30	45	45
TOTAL	580	980	775	1,145
7. CARGO BAY OPERATIONS	SMALL SPACECRAFT - GOOD; LARGE SPACECRAFT - GOOD.		SMALL SPACECRAFT - INTERFERENCE PROBLEMS; LARGE SPACECRAFT - GOOD.	
8. SERVICER OPERATING LENGTH (in.)	TUG	ORBITER	TUG	ORBITER
STOWAGE RACK	40	40	40	40
SERVICER MECHANISM	156	156	8	8
DOCKING MECHANISM	(DOES NOT ADD LENGTH)		50	50
TUG ADAPTER	0	N/A	0	N/A
SUPPORT STRUCTURE	N/A	8	0	8
TOTAL	196	204	98	106
9. SERVICER STOWED LENGTH (in.)	TUG	ORBITER	TUG	ORBITER
STOWAGE RACK	40	40	40	40
SERVICER MECHANISM-STOWED	94	94	8	8
DOCKING MECHANISM	0	0	50	50
TUG ADAPTER	0	N/A	0	N/A
SUPPORT STRUCTURE	N/A	8	N/A	8
TOTAL	134	142	98	106

2. Spacecraft Location and Distribution of Modules

The pivoting arm servicer allows access to modules on one complete end of the spacecraft. This can be two ends if the servicer is docked a second time. The general purpose manipulator servicer allows access to the outer surface of the spacecraft. The current design has a two-tier configuration. The spacecraft surface access is illustrated in Figure IV-14. The shaded points or areas represent the areas where modules can be located. When it is realized that the spacecraft will be configured on a 15-foot diameter to effectively utilize the orbiter cargo bay, the pivoting arm servicer must be rated as having the more effective module pattern and distribution. It allows greater flexibility for mounting modules in the central region as shown in Figure IV-15. This relates to flexibility in module shapes. A significant advantage of the pivoting arm end access is the flexibility in grouping of spacecraft subsystems in a given module. The spacecraft designer can group subsystems which require high heat dissipation together or keep them in separate modules according to what is dictated by good design. These modules can then be located on the spacecraft outer surfaces for efficient heat radiation to space. The subsystems which do not require heat dissipation can be mounted in the central region as separate modules. The keystone-shaped modules of the general purpose manipulator do not accommodate very effectively to this spacecraft design factor.

3. Module Sizes and Shapes

The pivoting arm servicer has a significantly greater flexibility than the general purpose manipulator in the area of module sizes and shapes. This mainly results from the problem the general purpose manipulator concept has with access to the central region of the spacecraft. To have good volumetric efficiency the general purpose manipulator dictates modules shaped similar to pie sections or keystone shaped sections. This is restrictive to the spacecraft designer. However, the pivoting arm servicer will accommodate any size of module up to the largest that the servicer stowage rack has a space for. The pivoting arm servicer does have the disadvantage of one tier of modules, but this can be handled by a secondary docking at the opposite end of the spacecraft.

4. Volumetric Efficiency

The definition of volumetric efficiency which is applicable evolves from the cost effective shape of the spacecraft dictated by the size of the orbiter cargo bay. Since launch costs are to be based on length in bay utilized and weight, it becomes important for the spacecraft designer to effectively utilize the 15-foot diameter of the cargo bay. This is a change to the designer because current launch vehicles are considerably smaller in diameter. Thus the spacecraft designer must work toward a vehicle envelope which is the conventional basic cylindrical, but it is larger in diameter and shorter in length than current conventional standards. The pivoting arm servicer very effectively accommodates use of the central region, resulting in efficient packing of subsystems and good volumetric efficiency. The short length also indicates a one-tier set of modules will probably be adequate which tends to negate one feature of the general purpose manipulator.

5. Mechanization Complexity - Degrees of Freedom

The general purpose manipulator servicer is relatively complex as compared to the pivoting arm because of the differences in number of degrees of freedom and in the number of linear drives involved in the design. There are four degrees of freedom plus a 180 degree index capability in the pivoting arm servicer design. This consists of three rotary joints and one linear travel in the wrist. The general purpose manipulator has six degrees of freedom: five rotary joints and one linear. The rationale for why linear travel joints are ranked more complex than rotary was given in the previous section. Two more degrees of freedom does result in a more complex design rating for the general purpose manipulator. Also, the additional linear drive requires a circular track wrapped around the periphery of the servicer. This is a complex type of drive both from complexity as well as ability to maintain alignment tolerances for exchange of modules during the servicing operation.

6. Servicer Weight

The servicer weights shown in Table IV-19 have been divided into the tug and orbiter classes because of the different equipment involved in each case. These weights are estimates in some cases because of insufficient definition in the literature. The servicer mechanism and stowage rack would

be the same obviously for each case. The structural design would be based on the most severe loads which would be the orbiter crash loads.

The general purpose manipulator servicer mechanism weights 150 lbs more than the pivoting arm servicer mechanism because of the heavy circular gear track in the general purpose manipulator design. The stowage rack weight of 400 lbs has been taken as the same for all configurations to keep the comparison more equitable.

The tug adapter is required to interface the servicer to the tug for servicing of high and medium earth orbit spacecraft. The servicer would be fastened directly to the front end of the tug during launch. The general purpose manipulator adapter weighs more because it must support a heavier servicer mechanism. The support equipment (450 lbs) listed in the table for orbiter application is for operation of the servicers out of the orbiter cargo bay. It supports the stowage rack and servicer mechanism. The control electronics assembly for the general purpose manipulator is 15 lbs more than for the pivoting arm because of the delta required in control logic due to the greater number of degrees of freedom.

7. Cargo Bay Operations

The operation of the pivoting arm and general purpose manipulator on-orbit servicers in the cargo bay for low earth orbit servicing was investigated. Since the size of the spacecraft has a major effect, the effort was divided into small and large spacecraft.

Small Spacecraft - Illustrations of small spacecraft being serviced in the orbiter's cargo bay are shown in Figures IV-17 and 18. If appendages do not interfere, the SRMS can dock the spacecraft to the servicer mechanism as shown. This approach would work very well for the pivoting arm servicer because it can function within the cargo bay's lower half envelope. However, the general purpose manipulator could only service half of the spacecraft due to interference with the cargo bay. The problem can be alleviated by mounting the spacecraft on a turntable or redocking it with the SRMS.

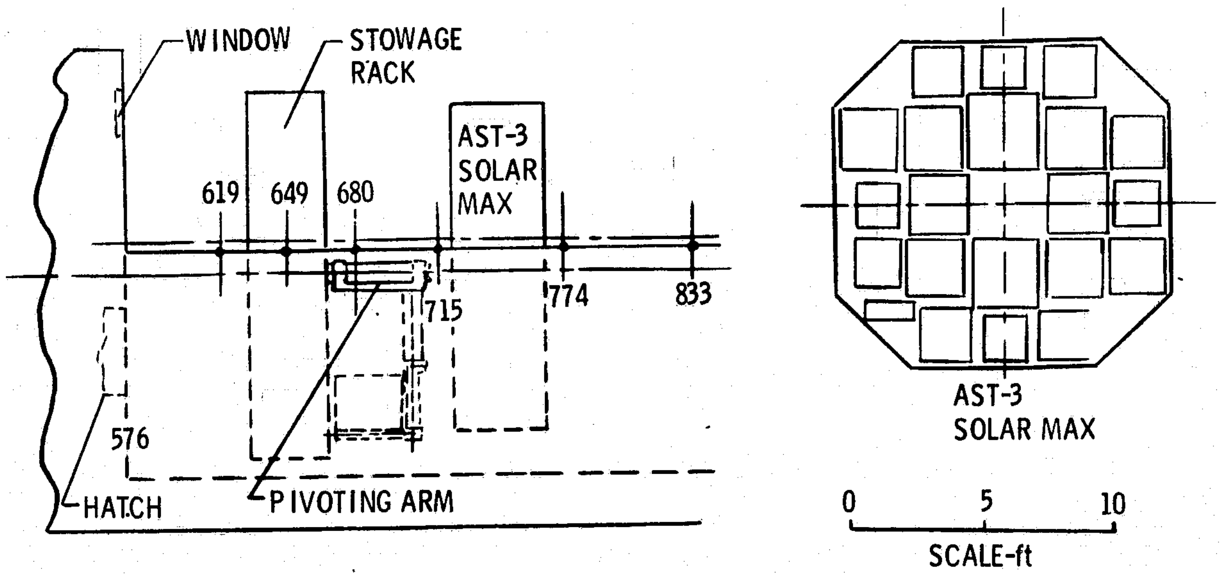


Figure IV-17 Pivoting Arm (TRW) - Shuttle Cargo Bay - Small Spacecraft

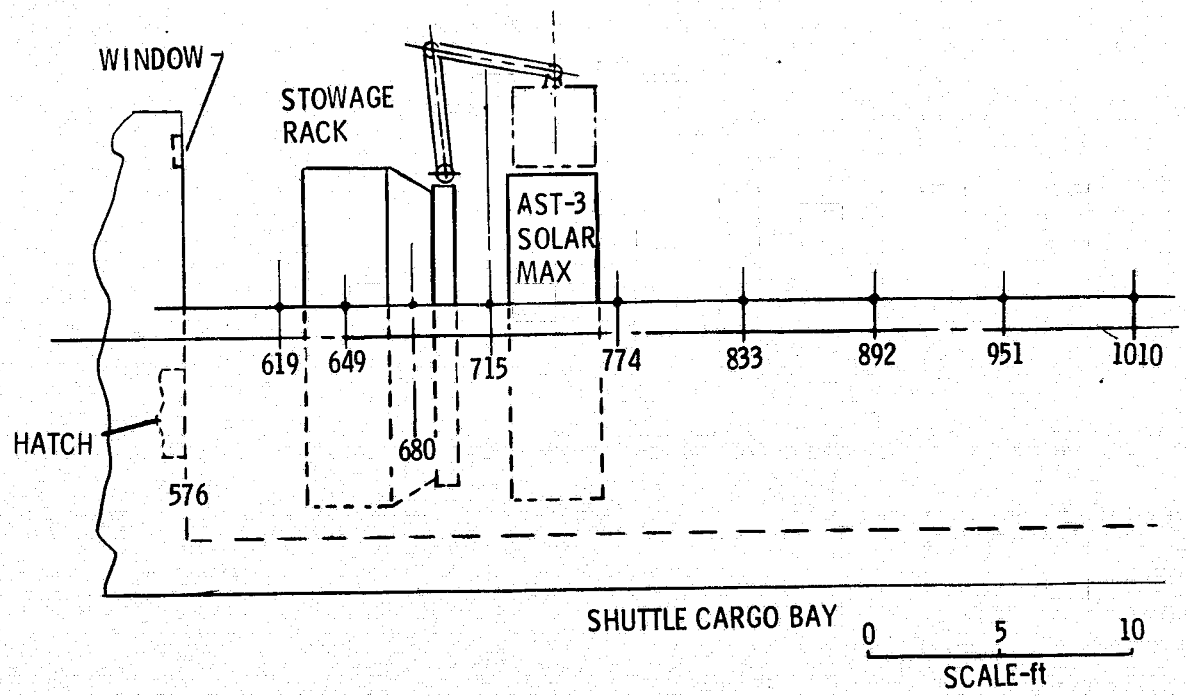


Figure IV-18 General Purpose Manipulator (MDAC)-Shuttle Cargo Bay-Small Spacecraft

Large Spacecraft - Both servicers work well with large spacecraft which, by their nature, must be docked outside of the cargo bay envelope. Figure IV-19 illustrates a representative servicing configuration of the pivoting arm servicer. The general purpose manipulator servicer would have a similar configuration.

8. Servicer Operating Length

The operating length of the servicer is defined as including the stowage rack and runs up to the servicer/spacecraft interface. Table IV-19 shows a detailed breakdown of the elements which go to make up the operating length.

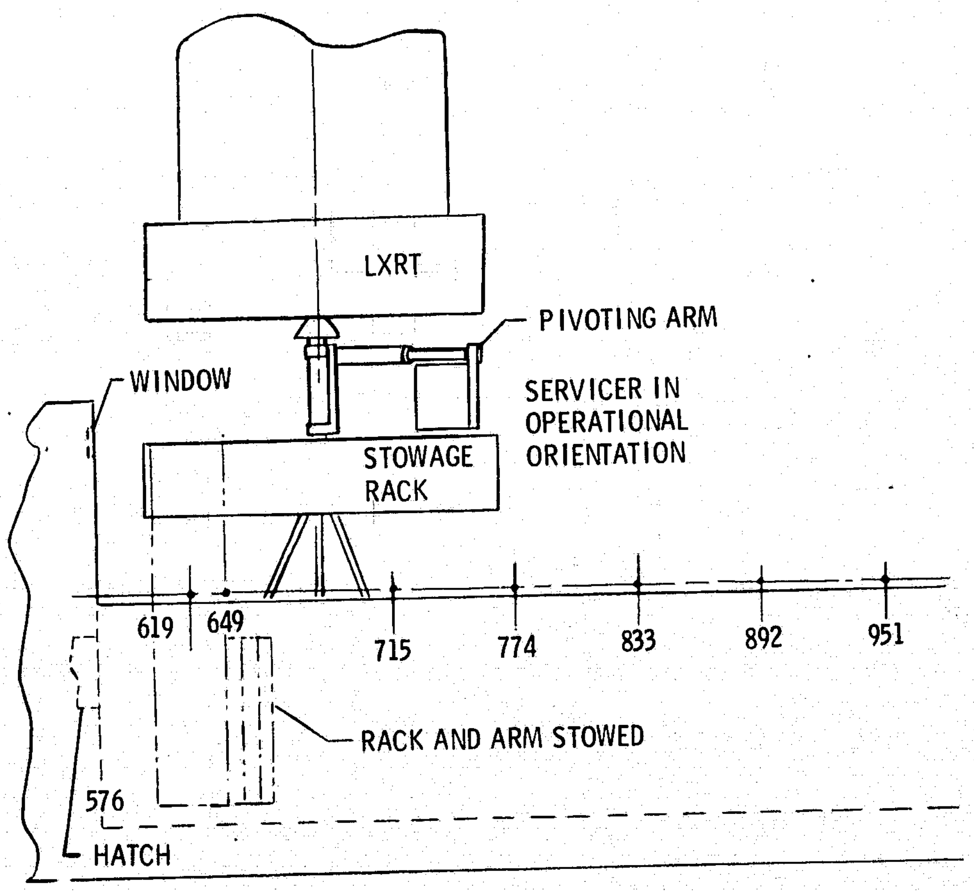


Figure IV-19 Pivoting Arm (TRW)-Shuttle Cargo Bay-Large Spacecraft

A fixed stowage rack length of 40 inches was used for both concepts. Some estimates had to be made because of insufficient information in the literature. Figures were compiled for tug and orbiter applications. The orbiter application requires a support structure which attaches to the sills in the cargo bay. The general purpose manipulator requires 98 inches less operating length because the manipulator moves around the outside whereas the pivoting arm operates between the spacecraft and stowage rack.

9. Servicer Stowed Length

Since launch costs are based on stowed length, it is an important parameter. The details on the stowed length comparison are shown in Table IV-19. The general purpose manipulator servicer stows in 36 inches less than the pivoting arm servicer. This is a definite advantage.

10. Summary of Pivoting Arm (TRW) and General Purpose Manipulator (MDAC) On-Orbit Servicer Comparison

A top level summary of this comparison is presented in Tables IV-20 and 21. The check marks in the tables show that both servicers are technically

Table IV-20 Technical Considerations

	TRW	MDAC
CRITERIA		
SATISFY ALL DESIGN CRITERIA	X	X
TECHNICAL FEASIBILITY	X	X
LIGHEST	X	
SHORTEST STOWED LENGTH		X
SIMPLEST	X	
OTHER CONSIDERATIONS		
OPERATIONAL AREAS		
LEO	X	X
HEO/MEO	X	X
PROGRAMMATIC ITEMS		
EXTENDABLE TO COMPLEX SERVICING TASKS		X

Table IV-21 On-Orbit Servicer Versatility

	TRW	MDAC
SPACECRAFT DESIGN ITEMS		
COMPATIBLE WITH A HIGH PERCENTAGE OF SPACECRAFT PROGRAMS	X	X
LEAST CONSTRAINTS ON NUMBER, LOCATION, SHAPE, SIZE, AND ORIENTATION OF MODULES	X	
CAPABLE OF AXIAL AND RADIAL MODULE REPLACEMENT	--	--
(INITIAL USE OF EITHER MODE WITH GROWTH TO THE OTHER IS ACCEPTABLE)	X	X
CAPABILITY FOR MODULE LOCATION IN MULTIPLE TIERS	X	X
CAPABILITY TO MAXIMIZE VOLUMETRIC EFFICIENCY	X	
COMPATIBLE WITH SIDE AND BOTTOM MOUNTED LATCHES	X	X
SPACE TRANSPORTATION SYSTEM DESIGN ITEMS		
ADAPTABLE TO A RANGE OF DOCKING DEVICES		
CENTRAL	X	X
PERIPHERAL	X	X

feasible which certainly is a valid conclusion. However, the previous discussion has detailed how the pivoting arm servicer is somehwat better in many of the comparison factors. Considering the differences in all comparison factors the pivoting arm design rates much better than the general purpose manipulator and is recommended to be carried forward as the best on-orbit servicer design found in the literature. However, it is also recommended at this stage in the study that the stowed and operating lengths for the pivoting arm be investigated. Design factors affecting these parameters should be evaluated to determine methods of minimizing these parameters.

D. COMPARISON OF VISITING MAINTENANCE SYSTEMS

In this section, a final comparison of the visting systems (Figure IV-20) is made. The visting systems at this point in the evaluation are three classes: (1) on-orbit servicer, (2) extravehicular activity (EVA), and (3) shuttle remote manipulator system (SRMS). Tradeoffs of on-orbit servicers done in other sections have resulted in the selection of the pivoting arm servicer as being the most effective design. Thus, in this section the EVA and SRMS maintenance systems will be compared to the pivoting arm which represents the most effective on-orbit servicer.

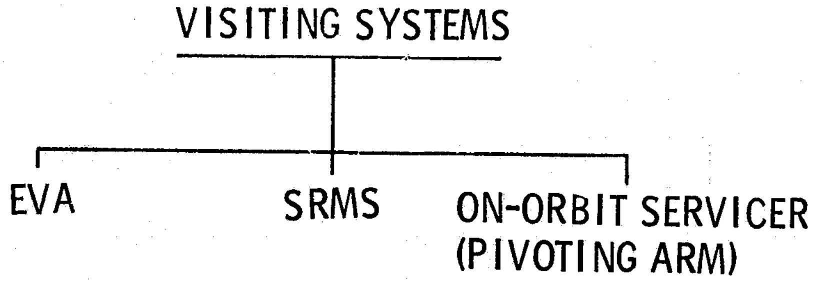


Figure IV-20 Visiting Systems

The major objective of this section is to determine the relative technical ranking of on-orbit servicer, EVA, and SRMS as visiting maintenance concepts for performing modular spacecraft servicing. This requires defining configurations which are representative but not necessarily an optimum choice. The definitions of the configurations are carried to a sufficient level of detail to establish technical feasibility and to provide an adequate breakdown of equipment required for a realistic costing analysis which is detailed in Chapter IX, Cost Generation and Analysis.

The most significant conclusions drawn in the analyses of this section are:

1. The on-orbit servicer maintenance concept is recommended as being the most effective,
2. The on-orbit servicer, extravehicular activity, and shuttle remote manipulator system are all technically feasible,
3. Only the on-orbit servicer is applicable to both tug and orbiter-based missions,
4. Design of the spacecraft for EVA is an important factor to consider in the cost analyses,
5. The additional support structure necessary for large spacecraft for EVA and SRMS maintenance requires a large stowage volume which will have a concurrent launch cost penalty,
6. The addition of a module exchange capability to the SRMS represents a significant increase in its design requirements and accuracy, and will result in a cost impact.

The ground rules used in the evaluation are summarized in Table IV-22. As was previously stated, the pivoting arm represents the on-orbit class of

Table IV-22 Visiting System Comparison Ground Rules

USE PIVOTING ARM AS REPRESENTATIVE OF ON-ORBIT SERVICERS.
SPACECRAFT ARE DESIGNED FOR SERVICING.
CONSIDER MODULE LEVEL REPLACEMENT AS BASELINE.
SERVICING IS TO BE PERFORMED IN THE ORBITER CARGO BAY.
USE THE JSC SPACE SHUTTLE SYSTEM PAYLOAD ACCOMMODATIONS DOCUMENT (JULY 3, 1974).
SRMS WILL RETRIEVE, DOCK AND DEPLOY THE SPACECRAFT FOR ALL MAINTENANCE CONCEPTS.
EVALUATE THE MAINTENANCE CONCEPTS INDEPENDENTLY (NOT IN COMBINATION).
FOR ALL MAINTENANCE CONCEPTS CONSIDER THE SPACECRAFT TO BE DOCKED TO SUPPORT STRUCTURE MOUNTED IN THE ORBITER CARGO BAY.
USE SAME REPLACEMENT MODULE STORAGE RACK FOR ALL CONCEPTS (ITS SUPPORT STRUCTURE CAN VARY).
ALL MODULES ARE LOCATED ON ONE OR TWO SEPARATE DOCKING FACES OR TWO ADJACENT TIERS.
APPENDAGES ARE TO BE NONRETRACTABLE.
APPENDAGES ARE ASSUMED TO HAVE LONG LIFE AND HIGH RELIABILITY AND THEREFORE DO NOT NEED REPLACING.

servicers. Module level replacement is baseline for all the concepts with capability to vary from this cited as an advantage when applicable. The JSC Space Shuttle Payload Accommodations Document (Chapter XI, Item D-2) has been used as the latest reference for EVA and SRMS design and operational data. The STS operational procedure for retrieving, docking and deploying of spacecraft is to use the SRMS. This is the method used across all three maintenance concepts. No combination of the three maintenance concepts was evaluated. However, when major advantages could be gained through a combination of concepts, these items were cited. The spacecraft is docked to a support structure mounted to the cargo bay sills or to the servicer mechanism via a docking interface. In the comparison, the stowage rack was kept the same for all concepts to eliminate minor differences in this area. Appendages were

considered to be nonretractable. This interrelates with the manner in which a spacecraft can be docked in the cargo bay.

In this comparison of the visting systems, the approach taken was to work the following areas for each system and perform a comparison across systems of the common factors.

1) Establish servicing guidelines,
2) Select representative servicing concept(s),
3) Determine acceptable operating region in the orbiter cargo bay,
4) Determine equipment required to perform servicing --
    - servicing concept equipment,
    - spacecraft equipment,
    - STS equipment, and
5) Determine advantages and disadvantages.

The technical comparisons of the visting systems were then made on the basis of:

1) Acceptable techniques,
2) Operating regions in orbiter cargo bay,
3) Equipment required, and
4) Significance of advantages and disadvantages.

1. Servicing Guidelines

Guidelines to be factored into the evaluation of each visiting system for formulating comparison factors were generated and a discussion of them follows.

Pivoting Arm Servicing Guidelines - The evaluation of the pivoting arm servicer was based primarily on current literature definition available, and was adapted to a servicer operation in the orbiter cargo bay. The pivoting arm servicing guidelines are listed in Table IV-23. Presently, the STS program has not established through a standard type control document any fundamental guidelines for the pivoting arm type of servicing operation in the orbiter cargo bay.

EVA Servicing Guidelines - EVA operational guidelines (Table IV-24) have been abstracted from the JSC Space Shuttle System Payload Accommodations document (July 3, 1974). A considerable number of EVA requirements have been put down as STS program control factors. However, a very significant factor to

Table IV-23 Pivoting Arm Servicing Guidelines

EVALUATION IS BASED PRIMARILY ON LITERATURE DEFINITION OF THE SERVICER.
ADAPTATIONS OF LITERATURE DEFINITION ARE APPROPRIATE.
NO CONTROL SYSTEM CONSTRAINTS.
COMPATIBLE WITH BOTH TUG AND ORBITER OPERATIONS.
SRMS IS USED FOR EXCHANGE OF MODULES NOT REACHABLE BY SERVICER DURING SERVICING AT ORBITER.

Table IV-24 EVA Servicing Guidelines

TWO, TWO-MAN, SIX-HOUR EVAs PER MISSION
PRE-EVA OPERATIONS - 3.5 hr
POST-EVA OPERATIONS - 1.5 hr
PLANNED EVA PERIODS SHOULD NOT EXCEED ONE 6-hr DURATION PER DAY
EVA CREWMAN AND EQUIPMENT MUST BE SECURED OR TETHERED AT ALL TIMES
CAPABILITY FOR CREWMAN TRANSLATION FROM AIRLOCK TO WORK AREA MUST BE PROVIDED
THE FIXED MANIPULATOR ARM MAY BE USED TO PROVIDE A TRANSLATION PATH
EITHER ONE AND/OR TWO CREWMEN MAY BE CONSIDERED FOR EVA OPERATIONS
PAYLOAD EQUIPMENT OR SURFACES SENSITIVE TO PHYSICAL DAMAGE SHOULD BE PROTECTED OR NOT LOCATED IN WORK AREAS

note is that the requirements are for general EVA operations in the orbiter bay and no specific servicing requirements exist. This means that even though EVA is certainly baseline to the STS program, EVA servicing of spacecraft has not reached a unified common approach which could guide the analyses conducted here.

EVA operations will utilize a self-contained life support system capable of supporting a six-hour EVA. Currently, three hours of prebreathing is required because of the 4 psi suit pressure. Considerations are being given to changing to an 8 psi suit pressure which would not require the prebreathing. The payload is charged with two of the three prebreathing hours plus 0.5 hour in the airlock.

The payload bay EVA support equipment available for all missions has not been defined at this time. If additional support equipment is required by a payload, its weight will be chargeable to the payload. When the manipulator end-effector is secured to the worksite, it may be used to provide a translation path.

SRMS Servicing Guidelines - SRMS operational guidelines (Table IV-25) have been abstracted from the JSC Space Shuttle System Payload Accommodations document (July 3, 1974). The shuttle remote manipulator system has been proposed as a candidate for servicing spacecraft docked in the orbiter cargo bay. However, the design has been driven by the requirements imposed from the operations of spacecraft retrieval and docking. No SRMS module exchange servicing approaches are baselined in the payload accommodations document. One manipulator arm is baselined. A second arm can be installed if required, but the weight (approximately 700 lbs) is chargeable to the payload. A basic jaw-type end-effector is currently baselined. Other end-effectors which probably would be more compatible with servicing requirements are being considered. TV cameras located in the bay and on the manipulator arm give the manipulator controller views that are selectable in viewpoint and field of view. A significant SRMS factor to be considered for the servicing application is its limitation in reaching around large objects. Also, this capability varies as a function of the location and orientation of the spacecraft in the cargo bay.

2. Visiting Systems Servicing Concepts

Pivoting Arm Servicer - A description of the pivoting arm servicer has been discussed in detail in the previous sections. The servicer has versatility in its application to both tug and orbiter servicing operations. One difference exists in the design for the two applications. The adapter used to interface the stowage rack to the tug or orbiter must be different. The

Table IV-25 SRMS Servicing Guidelines

NO MODULE EXCHANGE EXISTS AS SRMS BASELINE DEFINITION
ARM DESIGN DRIVEN BY SPACECRAFT RETRIEVAL AND DOCKING REQUIREMENTS
ONE ARM IS BASELINE
SECOND MANIPULATOR ARM CAN BE INSTALLED IF REQUIRED
END EFFECTOR - BASIC JAW TYPE
LIMITED CAPABILITY TO REACH AROUND LARGE OBJECTS
ONE ARM CLEAR REACH ENVELOPE - HEMISPHERE ABOVE CARGO BAY WITH RESTRICTED REGIONS DUE TO SHOULDER TWO DEGREES OF FREEDOM
WRIST MINIMUM LATERAL FORCE - 10 lb
DIRECT VIEW - AVAILABLE BUT LIMITED
INDIRECT VIEW - SELECTABLE IN VIEWPOINT AND FIELD OF VIEW
DEGREES OF FREEDOM - 6

pivoting arm servicer operating in the orbiter cargo bay is illustrated in in Figure IV-21. The spacecraft docking interface is the same as it is for the tug application. The SRMS docks the spacecraft to be serviced to the front of the servicer mechanism on the docking mechanism. Prior to the spacecraft docking several steps must be performed to ready the servicer for the docking operation. The servicer (stowage rack, servicer mechanism, and docking interface) are launched as an integral structure supported from the sills and returned in the cargo bay in a standard cylinder type orientation. To prepare for servicing, the SRMS removes the servicer from the stowed location which can be anywhere in the bay that the SRMS can reach. The sill automatic latch releases are used for this because they can be remotely controlled from the crew control station. Then the SRMS places the servicer in a vertical orientation (Figure IV-21) at the desired location in the bay. The automatic sill latches are again used for holding the servicer in position. The location is flexible. It is restricted only by other cargo in the bay and the SRMS reach capability. Then once the spacecraft has been

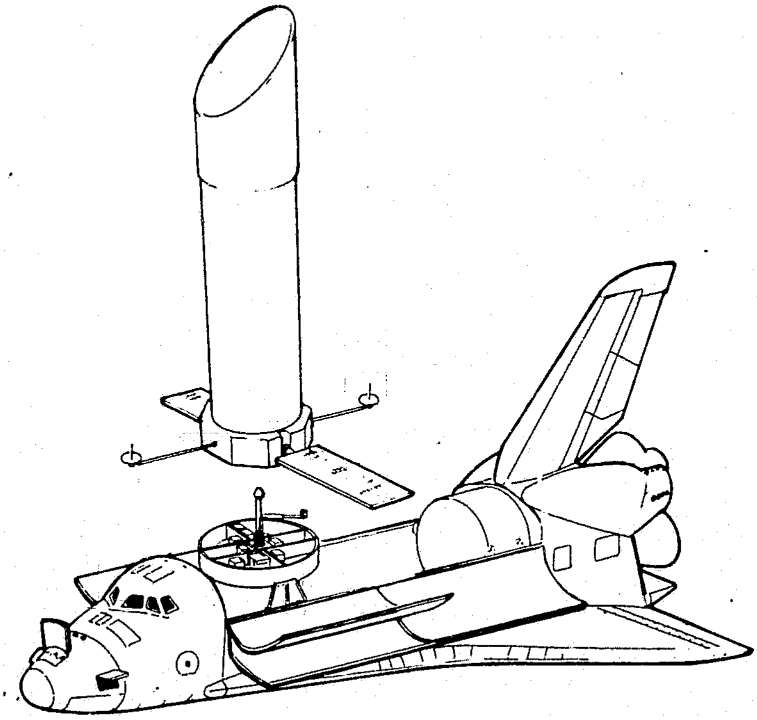


Figure IV-21 Pivoting Arm Servicing in Orbiter Cargo Bay

docked by the SRMS, module exchange takes place in the same manner that it does for the tug application with possibly an exception on the servicer control mode. If the servicing can take place near the forward end of the cargo bay as shown in Figure IV-21, then the operation could be controlled with the monitoring being done through the rear facing windows and the cargo bay TV cameras.

EVA Servicing Concept - Considerable EVA contingency type servicing was performed successfully during the Skylab program. This data substantiates that a wide range of EVA servicing tasks can be done by a suited astronaut. However, the current space shuttle payload accommodations document does not provide baseline EVA approaches. A survey of the literature shows that various EVA servicing approaches have been proposed. None of them presents an approach which is based on a servicing requirement for accommodating spacecraft which vary considerably in size and configuration. Two approaches representative of EVA servicing for large and small spacecraft are shown in Figure IV-22. It is important to note that the type of EVA servicing that can be done will be governed by the limited dexterity the crewman has in a pressurized suit. Conventional handrails and tethers

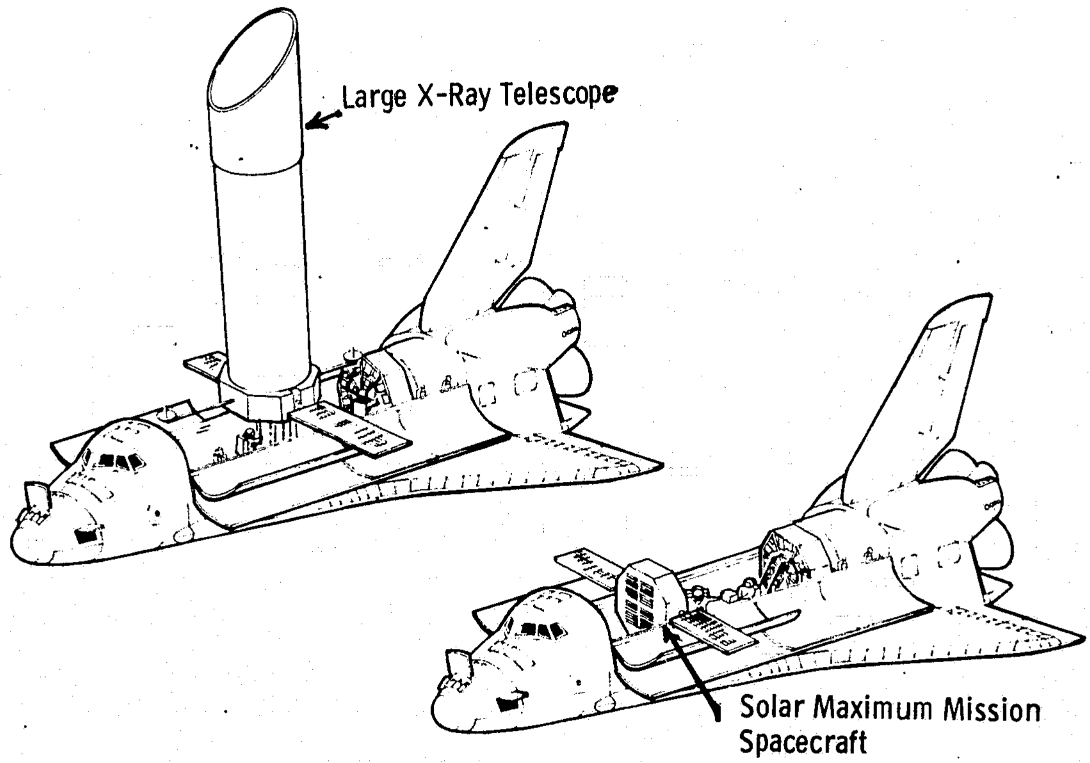


Figure IV-22 EVA Servicing Concept

will have to be provided. EVA design, or what is sometimes called man-rating, of the spacecraft to permit the crewman to work on or near the spacecraft represents a potential cost impact factor. Another significant factor is the support structure required to support large spacecraft above the cargo bay. This structure would be mounted to the bay sills. Stowage volume and deployment of such a large structure are considerations to be assessed for cost impacts.

The EVA design elements considered were taken from Item M-18 of Chapter XI. They can be thought of in terms of providing a payload safe work station where many of the requirements are established by the shuttle program for everything carried in the orbiter cargo bay. The additional

requirements (per Item M-18) are 1) provision of EVA load bearing surfaces for hand/foot restraints and pushoff, 2) additional structural protection (contamination and thermal control) where orbital conditions differ from ground, and 3) secondary power and/or AC power protection. Further considerations are 1) contamination effects including suit leakage and water vapor from the suit thermal control system, 2) compatibility with EVA suit restrictions regarding reach, forces available, hand grip sizes, and restraints, and 3) unintended contact between suit and spacecraft including its appendages.

Shuttle Remote Manipulator System Servicing Concept - The manipulator arm will be used to capture and dock spacecraft at the orbiter cargo bay. Servicing with the SRMS is considered to start once the spacecraft is secured to a spacecraft support structure in the bay. Servicing of small and large spacecraft present very different geometric problems relative to reach around capabilities of the arm and spacecraft appendage interferences. Representative approaches for servicing large and small spacecraft are illustrated in Figure IV-23. The relative geometry problem of how the arm handles module exchange between the stowage rack and spacecraft is discussed later in this section.

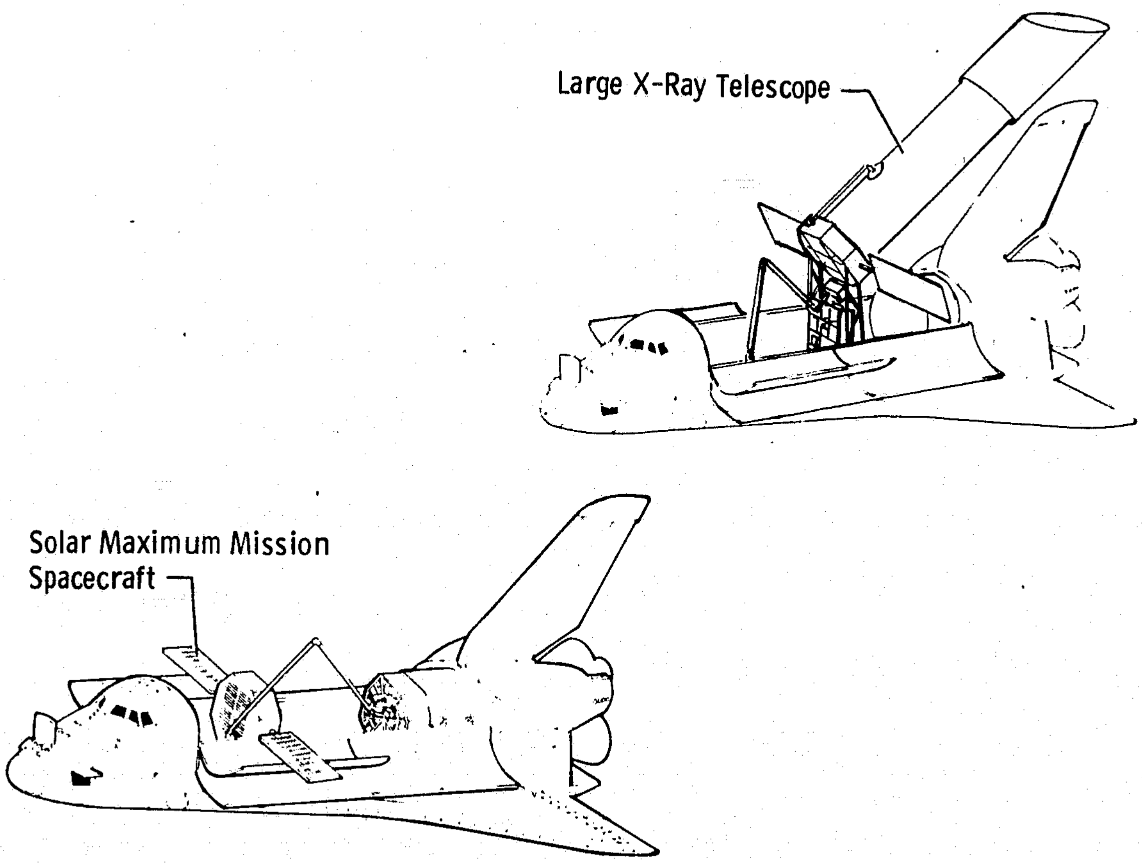


Figure IV-23 Shuttle Remote Manipulator Servicing Concept

The design of the manipulator arm was driven by the prime requirement to provide the capability for capture, docking, stowage and deployment of large, heavy payloads. This arm design imposes two main restrictions on SRMS when used for a servicing operation. One restriction is in the area of reaching around objects. The mounting location of the arm and the two degree-of-freedom shoulder joint (as opposed to three degrees) create a significant reach restriction problem. The other restriction is the positional accuracy of the arm. Currently, it is felt that the present design will not have the positional accuracy necessary to pick up modules and place them.

The concepts shown in Figure IV-23 indicate how spacecraft must be mounted and located in the cargo bay to provide manipulator arm access to the spacecraft servicing areas. To provide complete access to all surfaces, the spacecraft will probably have to be mounted on a turntable. A second arm would minimize this problem to a degree. To minimize operational time and the hazard avoidance problem, the arm should be programmed for automatic control between spacecraft servicing and stowage rack locations.

3. Servicing Operating Regions in Orbiter Cargo Bay

An important consideration in the comparison of an on-orbit servicer, EVA and SRMS for performing servicing on large and small spacecraft is the cargo bay operating regions in which each can function.

Pivoting Arm Servicer - One orientation of the pivoting arm servicer can accommodate both large and small spacecraft effectively. Figure IV-19 illustrates the cargo bay layout that was arrived at. It is shown at the forward end of the cargo bay and at a height were monitoring from the crew quarters' window can be utilized in the servicing operation. Aside from this advantage, the servicer could be located anywhere in the bay that the SRMS can place it and dock spacecraft to it. There is also the obvious interference with other spacecraft located in the bay. However, scheduling of the sequence of program operations on a particular flight could result in a lot of available space in the bay. However, the pivoting arm servicer does accommodate well to this type of a restriction if it exists.

EVA Servicing - Few restrictions on location of the stowage rack and spacecraft are imposed by using an EVA servicing approach (Figure IV-24). It is very flexible to this factor. There is a need for the astronaut to get around other 15-foot diameter spacecraft which might be in the bay and a need for clear access to the hatch to the orbiter airlock.

SRMS Servicing - The large manipulator appears at first level of investigation to be very flexible as far as reach capabilities in the cargo bay. However, as geometric layout drawings are made, it becomes obvious that the SRMS has a considerable number of reach access problems for a variety of reasons. A representative functional layout for servicing small spacecraft

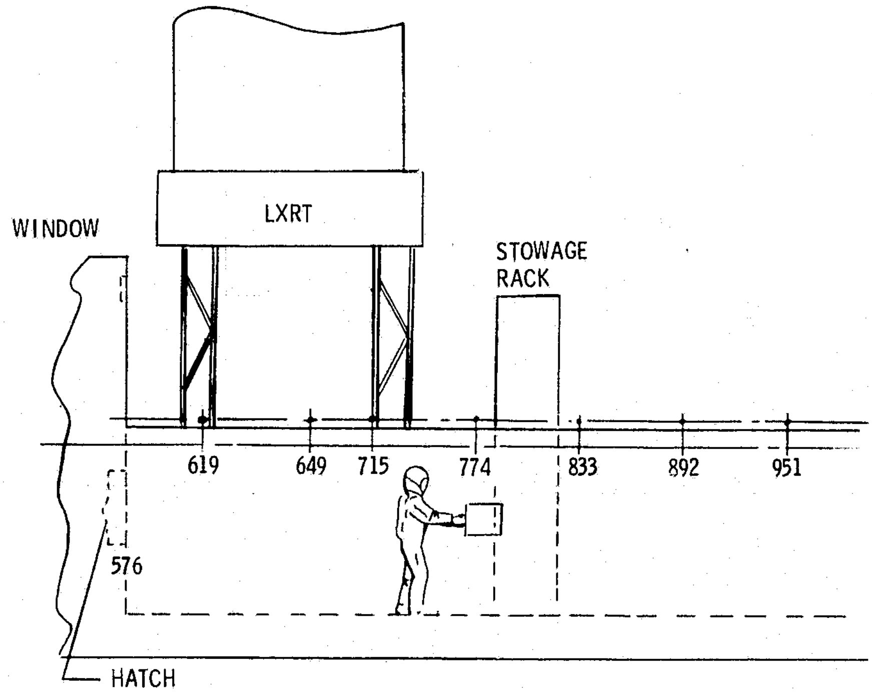


Figure IV-24 EVA Large Spacecraft

in the cargo bay is shown in Figure IV-25. For this type of orientation the arm can be seen to have interference with the stowage rack if it is moved further forward. As the stowage rack and spacecraft are moved more aft than shown, a point is reached where the arm once again interferes with the stowage rack. Details on this are summarized in a table later in this section.

Servicing operations with large spacecraft present different geometric locating problems in the bay than for small spacecraft. An effective configuration for SRMS servicing of large spacecraft like the LXRT is shown in Figure IV-26. The spacecraft is tilted 45 degrees aft from vertical out of the bay and is mounted to a support structure which is approximately 15 feet above the sills. This represents the most effective layout, but it has the disadvantage of requiring a very large support structure. The support structure must also be made so that it does not interfere with the manipulator removing modules. This is an effective layout for small spacecraft also, but it would not be practical to pay the penalty of the large support structure.

Summary of Servicer Operating Regions in the Cargo Bay - The dimensional details of the conclusions drawn on the operating regions for each of the

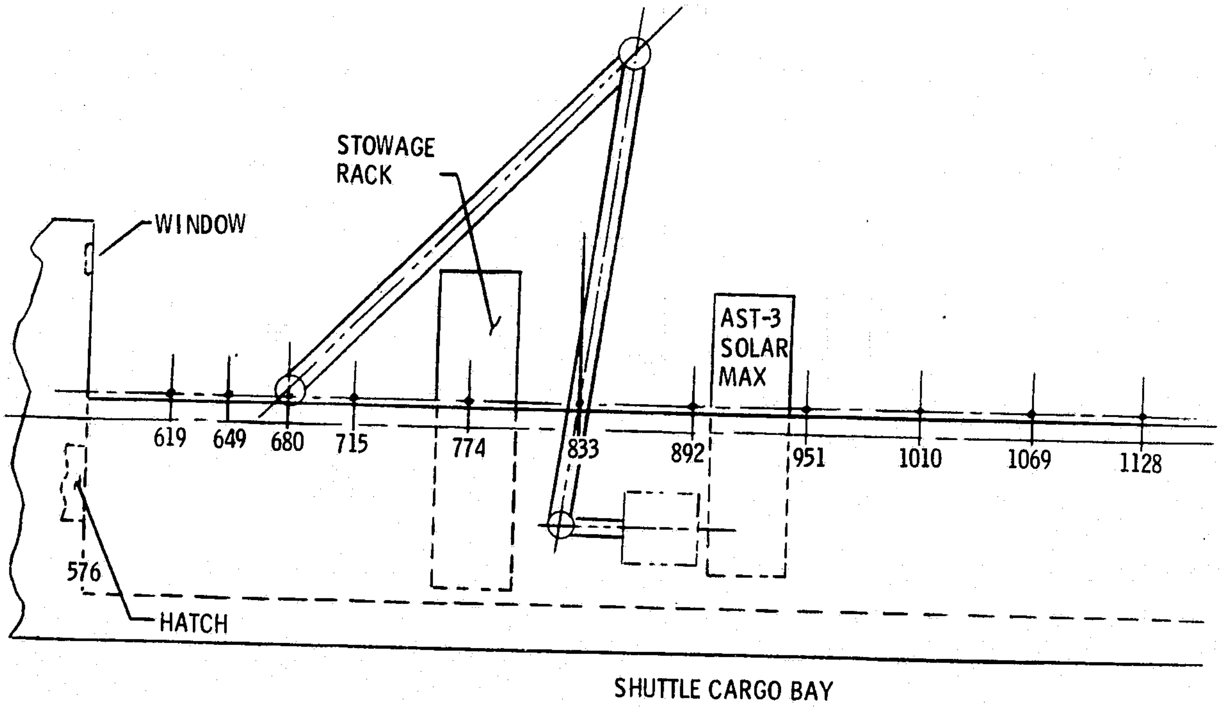


Figure IV-25 SRMS Small Spacecraft

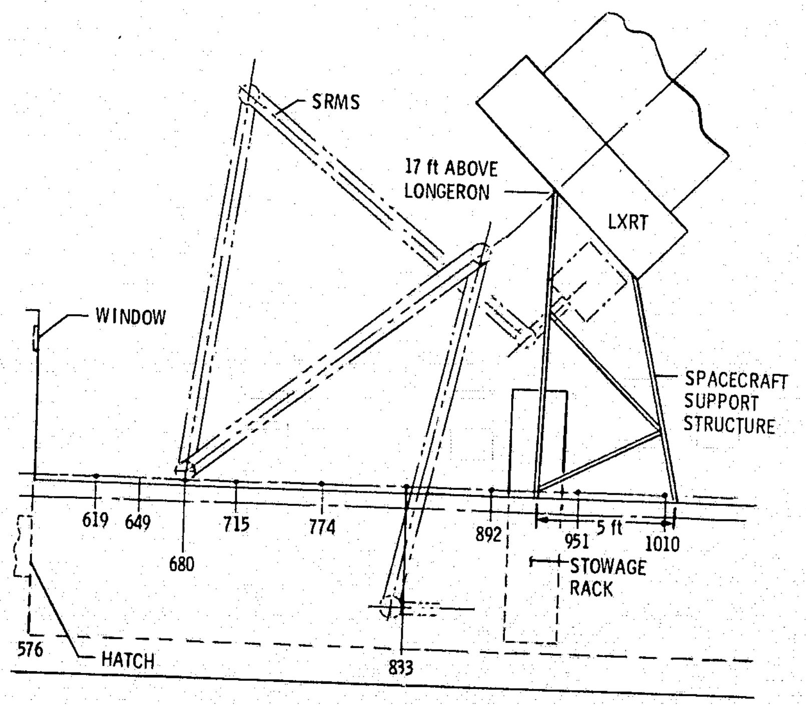


Figure IV-26 SRMS Large Spacecraft

three visiting systems are summarized in Table IV-26. The pivoting arm and EVA servicing result in no significant difficultues with respect to location in the cargo bay.

Table IV-26 Comparison of Operating Regions in Orbiter Cargo Bay

		EVA		SRMS			PIVOTING ARM	
SPACECRAFT SIZE		SMALL	LARGE	SMALL	LARGE PERPENDICULAR MOUNT	LARGE 45° MOUNT	SMALL	LARGE
DIRECTION OF MODULE REMOVAL		AXIAL PREFERRED	AXIAL PREFERRED	AXIAL PREFERRED	AXIAL OR RADIAL	AXIAL OR RADIAL	AXIAL ONLY	AXIAL ONLY
EVA HATCH CLEARANCE		NEEDED FOR ALL						
*AMOUNT OF CARGO BAY USABLE	SPACECRAFT	MOST OF BAY	MOST OF BAY (RESTRICTED BY APPENDAGES)	15 ft	AXIAL-36' RADIAL-29'	30 ft	MOST OF CARGO BAY USABLE	MOST OF CARGO BAY USABLE
	STOWAGE RACK			21 ft	AXIAL-46' AFT RADIAL-21' AFT	27 ft		
		(ALL ARE RESTRICTED SOMEWHAT BY WHERE THE SRMS CAN DOCK A SPACECRAFT)						
GENERAL CONDITIONS			MOUNT SPACECRAFT VERTICALLY		FOR RADIAL REMOVAL A ROTATABLE TABLE IS REQUIRED			
CONCLUSIONS		NO APPARENT DIFFICULTIES	NO APPARENT DIFFICULTIES	LIMITED USABLE SERVICING REGION	LIMITED USABLE SERVICING REGION. REQUIRES ROTATABLE SUPPORT STRUCTURE	LIMITED USABLE SERVICING REGION	NO APPARENT DIFFICULTIES	NO APPARENT DIFFICULTIES

*THE NUMBERS GIVEN ARE THE NUMBER OF FEET THE SPACECRAFT/STOWAGE RACK MUST BE MOUNTED AFT OF THE EVA HATCH.

4. Servicer Equipment Requirements

The servicer equipment requirements for each visiting system servicer have been listed in Table IV-27. The equipment required has been divided into three categories: (1) servicing concept, (2) STS, and (3) spacecraft. This has been done so the technical and cost impacts in each of the categories can be assessed. None of the equipment required poses a question of technical feasibility. STS baseline equipment is assumed to be used where applicable, e.g., docking mechanisms and aids. The cost impacts resulting from weight and stowage volume for the equipment are assessed in the cost analyses.

5. Visiting System Comparison Results

Summaries of the advantages and disadvantages for each of the three visiting system servicers are presented in Tables IV-28, 29, and 30. The

Table IV-27 Summary of Servicer Equipment Required

	PIVOTING ARM SERVICER	EVA	SRMS
SERVICING CONCEPT EQUIPMENT	MODULE STOWAGE RACK SERVICER MECHANISM INTERFACE WITH DOCKING DEVICE MODULE STOWAGE RACK ADAPTERS SPACECRAFT SUPPORT STRUCTURE (LEO)	MODULE STOWAGE RACK SPACECRAFT SUPPORT FRAME TRANSLATION AIDS AND TETHERS SIMULATION CREW TRAINING TIME MOCKUPS	SPECIAL PURPOSE END EFFECTOR SPACECRAFT SUPPORT FRAME (ROTATABLE) MODULE STOWAGE RACK ADDITIONAL ARM IF NEEDED SIMULATION CREW TRAINING TIME MOCKUPS
STS EQUIPMENT	DOCKING MECHANISM DOCKING SENSOR/AIDS LIGHTING } HEO	SHUTTLE REMOTE MANIPULATOR SYSTEM (DOCKING/DEPLOYMENT) DOCKING SENSORS AND AIDS LIGHTING SIMULATORS NEUTRAL BUOYANCY MOVING BASE	SHUTTLE REMOTE MANIPULATOR SYSTEM DOCKING SENSORS AND AIDS LIGHTING
SPACECRAFT EQUIPMENT	DOCKING MECHANISM DOCKING SENSORS/AIDS REPLACEABLE MODULES	DOCKING MECHANISM DOCKING AIDS HANDRAIL AND TETHER FASTENERS REPLACEABLE MODULES ACCESS TO MODULES CREW SAFETY DAMAGE-SENSITIVE SURFACE PROTECTION	DOCKING MECHANISM DOCKING AIDS REPLACEABLE MODULES ACCESS TO MODULES

Table IV-28 Pivoting Arm On-Orbit Servicing Summary

ADVANTAGES	DISADVANTAGES
APPLICABLE TO BOTH LEO AND HEO	SERVICING TASKS ARE LIMITED TO MODULE REMOVAL/REPLACEMENT
DEVELOPMENT FOR LEO AIDS GROWTH TO HEO	REQUIRES DEVELOPMENT OF A SERVICER MECHANISM
TOTAL MODULE EXCHANGE TIME IS LOW	MODULE REMOVAL DIRECTIONS LIMITED
EVA IS AVAILABLE FOR EVALUATING AND HANDLING CONTINGENCIES (LEO)	MINOR MODULE LOCATION RESTRICTIONS
SRMS IS AVAILABLE FOR OUTSIZE MODULE EXCHANGE (LEO)	
REQUIRES MINIMAL OPERATOR TRAINING	

Table IV-29 EVA Servicing Summary

ADVANTAGES	DISADVANTAGES
MODULES CAN BE LOCATED ANYWHERE ON SPACECRAFT SPACECRAFT CONFIGURATION IS NOT RESTRICTED MINIMUM DEVELOPMENT IS REQUIRED (SKYLAB EXPERIENCE) MAN IS AVAILABLE ON LOW EARTH ORBIT SHUTTLE MISSIONS MAN HAS ABILITY TO PERCEIVE ABNORMAL CONDITIONS AND COMPENSATE FOR THEM MODULE/LATCHES CAN BE COMPATIBLE WITH A HEO SERVICER TOTAL EVA TIME REQUIRED IS WITHIN BASELINE LIMITS (2.5 hr REQUIRED) MAN CAN DO BROADER RANGE OF SERVICING ACTIVITIES THAN MODULE EXCHANGE UTILIZES EXISTING SHUTTLE BASELINE EQUIPMENT	SPACECRAFT MUST BE DESIGNED FOR EVA PRESENT EVA DEVELOPMENT IS FEASIBLE ONLY TO SERVICE SPACECRAFT IN LEO CREWMAN SAFETY OPTICAL SURFACE AVOIDANCE TRAINING REQUIREMENTS SPACECRAFT MUST BE DESIGNED TO ACCOMMODATE ATTACHMENT OF TRANSLATION AND TETHER AIDS MECHANICAL FASTENERS MUST BE DESIGNED SPECIFICALLY FOR SUITED OPERATION EXTERIOR EQUIPMENT AND THERMAL CONTROL SURFACE PHYSICAL CONTACT AVOIDANCE TIME REQUIRED FOR PRE-EVA (3.5 hr) AND POST-EVA (1.5 hr)

Table IV-30 SRMS Servicing Summary

ADVANTAGES	DISADVANTAGES
UTILIZES EXISTING SHUTTLE BASELINE EQUIPMENT COST OF ONE ARM IS CHARGED TO THE SHUTTLE ORBITER SERVICING CAN BE PERFORMED ON MULTIPLE SURFACES OF THE SPACECRAFT GOOD FLEXIBILITY IN THE TYPES OF TASKS THAT CAN BE PERFORMED MAN IS AVAILABLE FOR: CONTINUOUS OPERATION OF ARM; MONITORING OF SERVICING STATUS; EVALUATING AND HANDLING CONTINGENCIES COMPATIBLE WITH MISSION SCHEDULE CONSTRAINTS ADAPTABLE TO BROADER RANGE OF SERVICING ACTIVITIES BY USING MORE DEXTEROUS END EFFECTOR	LIMITED TO SERVICING SPACECRAFT IN LOW EARTH ORBIT IF SECOND ARM IS REQUIRED, WEIGHT IS CHARGED TO THE PAYLOAD OPERATOR ERROR CAN BE DAMAGING TO THE SPACECRAFT RESTRICTS OPERATIONS OF SHUTTLE ORBITER DURING THE SERVICING ACTIVITIES TRAINING REQUIREMENTS

analyses of this section and an evaluation of the advantages and disadvantages for each visiting system have resulted in the following conclusions.

1. The on-orbit servicer maintenance concept is recommended as being the most effective.
2. The on-orbit servicer, extravehicular activity, and shuttle remote manipulator system are all technically feasible.
3. Only the on-orbit servicer is applicable to both tug and orbiter based missions.
4. Design of the spacecraft for EVA is an important factor to consider in the cost analyses.
5. The additional support structure necessary for large spacecraft for EVA and SRMS maintenance requires a large stowage volume which will have a concurrent launch cost penalty.
6. The addition of a module exchange capability to the SRMS represents a significant increase in its design requirements and accuracy, and will result in a cost impact.

V. SPACECRAFT INTERFACE ANALYSIS

The design of spacecraft so that they can be serviced on-orbit using a module exchange form of on-oribt servicer is an important aspect of orbital servicing. Each of the prior studies presented serviceable designs of their spacecraft so a good positive data base existed. The IOSS was not intended to delve deeply into specific serviceable spacecraft and thus did not. However, certain subjects were addressed to help develop our understanding of serviceable spacecraft.

The geometric interfaces between the spacecraft, modules, docking system, servicer mechanism, stowage rack, and carrier vehicle have been addressed. Part of the discussion is in Chapters IV and VIII and part is here. Consideration was given as to how the characteristic set might be configured for servicing and how the pivoting arm and general purpose manipulator forms of servicer mechanisms interact with the identified configurations. It was concluded that the IOSS is in agreement with prior studies that spacecraft can be designed to be serviceable with acceptable design, weight, and volume effects. The pivoting arm using axial module replacement was found to be compatible with serviceable spacecraft configurations that make greatest use of the STS capability.

A discussion of module characteristics is used to introduce module data from the Operations Analysis study and to show that modules need not be overly large or unduly small. Spacecraft composed of ten to 20 modules seem appropriate.

The important mechanical interface between modules and spacecraft (and stowage rack) was addressed significantly in the contract. That work is summarized here in the form of design criteria or guidelines, two specific SRU interface mechanism designs, an end effector design, and engineering test units of these items.

The influence of rendezvous, and more particularly docking, upon on-orbit servicing is discussed. There is much work to be done and emphasis should be given to convincing the user that rendezvous and docking will be operational and safe.

A. SPACECRAFT CONFIGURATIONS FOR SERVICING

The prior studies of Chapter II concluded that spacecraft could be designed to be serviceable. To have a study reference against which the pivoting arm and general purpose manipulators could be evaluated, the characteristic set of spacecraft were configured for servicing as representatives of the total maintenance applicable spacecraft set. As the BESS data was not available it was not included in this analysis. The configurations were to consider primarily module sizes, but with some consideration given to location requirements. It was found that, to a first level, spacecraft can be made serviceable without mission objective penalties and that the pivoting arm and general purpose manipulator servicers can accommodate a wide range of spacecraft characteristics without excessive configuration penalties.

The factors to be evaluated were:

1) Effect of modularizing the spacecraft on its size and shape,

2) Effect of larger module envelopes to allow for servicer access and removal clearance,

3) Effect of designing for "status quo" or "maximum STS efficiency,"

4) Effect of allowance for a docking mechanism, and

5) Comparison of configurations based on axial vs radial module replacement motion.

The conditions of the evaluations were:

1) Use of all spacecraft in the characteristic set except BESS,

2) Module densities 50% greater than those used in the Operations Analysis study (Chapter XI, Items F-19 and -20). This increase was based on the DSCS II work of TRW, the PUT work of Fairchild, and MMC experience,

3) Geometrical interfaces only were to be considered,

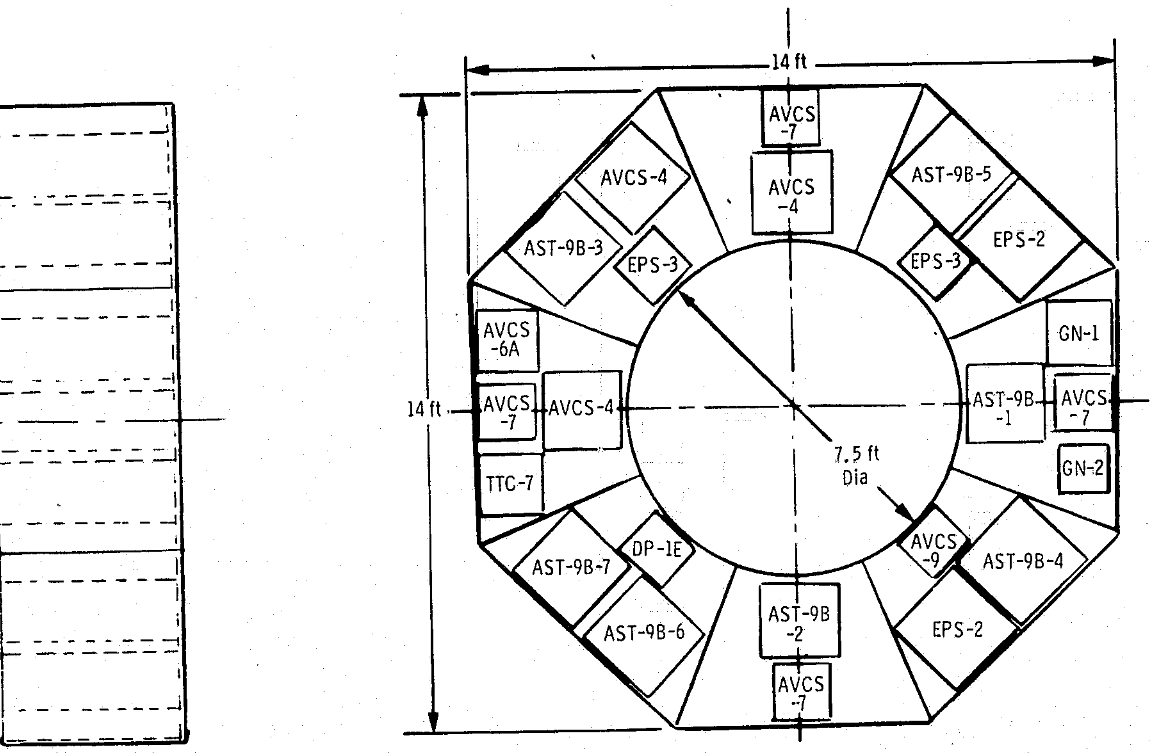
4) Single servicing missions on body stabilized spacecraft, and

5) The basic resources for orbital servicing were available (time, volume capacity, and weight capacity).

It became apparent that there are at least two configuration policies available which might be denoted as "status quo" and "maximum STS efficiency." "Status quo" configuration attempts to preserve expendable spacecraft envelopes and surface orientations while "maximum STS benefit" configuration strives to result in the most efficient use of the STS resources. Both are recognized; however, a preferred recommendation is not available at this time. The "status quo" approach recognizes that if servicing is possible with little overall envelope change or compromise to mission objectives, user acceptance may follow easily. On the other hand, the "status quo" configuration may not fully utilize the benefits and efficiencies of the STS. The "maximum STS benefit" approach does use all these potential capabilities. This latter approach may involve a greater impact on spacecraft design "tradition" affecting cost through overall management requirements, resource development and testing of spacecraft.

The expendable forms of the characteristic set of spacecraft are shown in Fig. I-1. Five of the six were configured as follows: The complement of modules was taken from the Operations Analysis (OA) work (Chapter XI, Items F-19 and 20) along with their weights. Where the OA modules were too small they were combined into larger modules. The OA weights were adjusted downward to allow for the heavy baseplates used by OA. The volumes were then computed using a set of densities representing best estimates from the DSCS II work of TRW, the PUT work of Fairchild, and MMC experience. Values ranged from 15 lbs/ft^3 for electronics to 35 lbs/ft^3 for electrical power systems. A representative upper bound on module length of 40 in. was selected based on cargo bay length considerations and some preliminary layouts. The access space was taken as four in. on the interface mechanism side and one in. on all other sides of each module.

The resulting configurations are shown in Figs. V-1 through V-5. The module designations are those of the Operations Analysis study except for Fig. V-3, the International Communications Satellite, which is an early configuration prepared by COMSAT Labs. The "status quo" concept of the UAE takes the form of two tiers of modules. It requires two docking interfaces for the pivoting arm servicer and one docking interface for the general purpose manipulator. It has a diameter more than half of the orbiter cargo bay diameter and thus does not store efficiently in the cargo bay. The "maximum



ure V-1 Large X-Ray Telescope

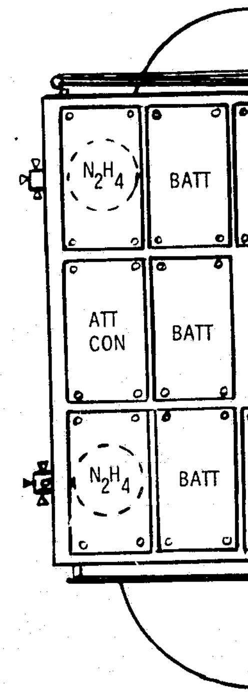
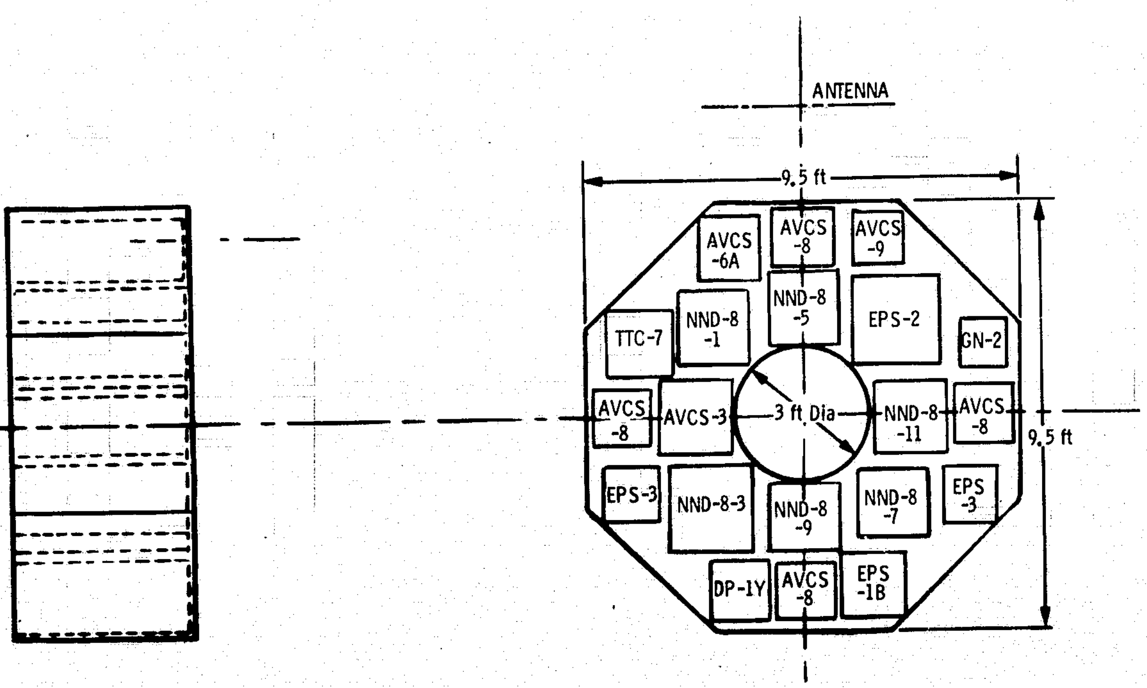


Figure V-3 In



re V-2 Environmental Monitoring Satellite

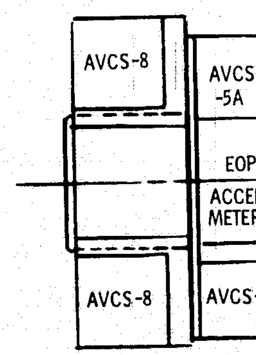
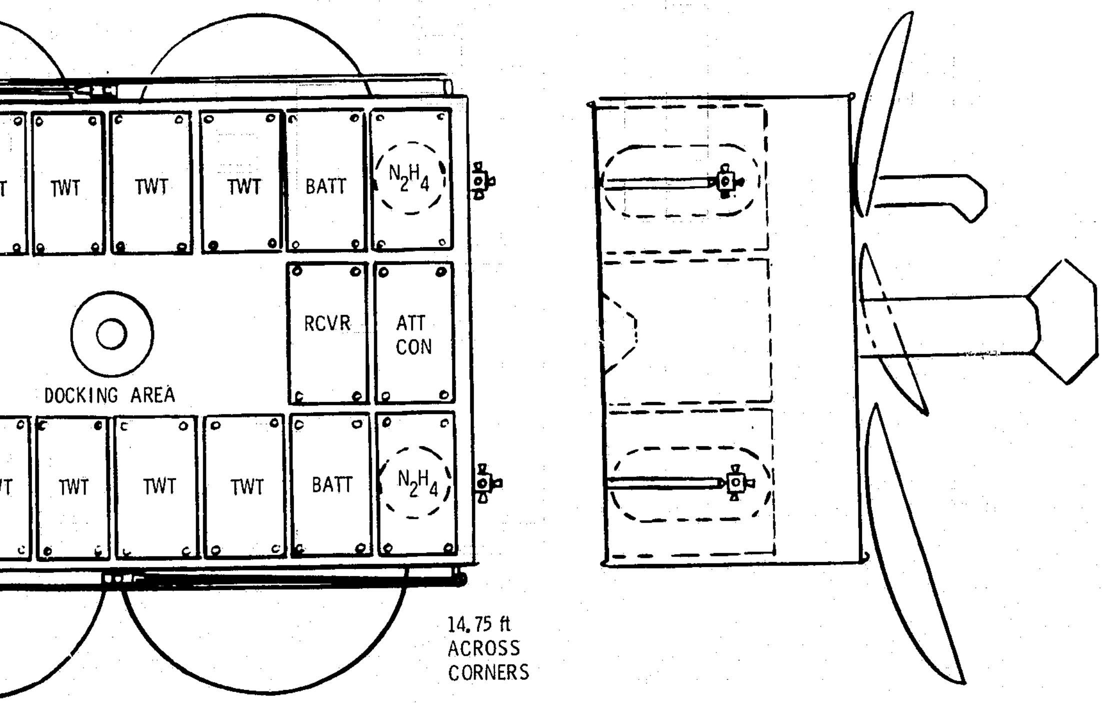


Figure V-4 Gr

FOLDOUT FRAME



national Communications Satellite

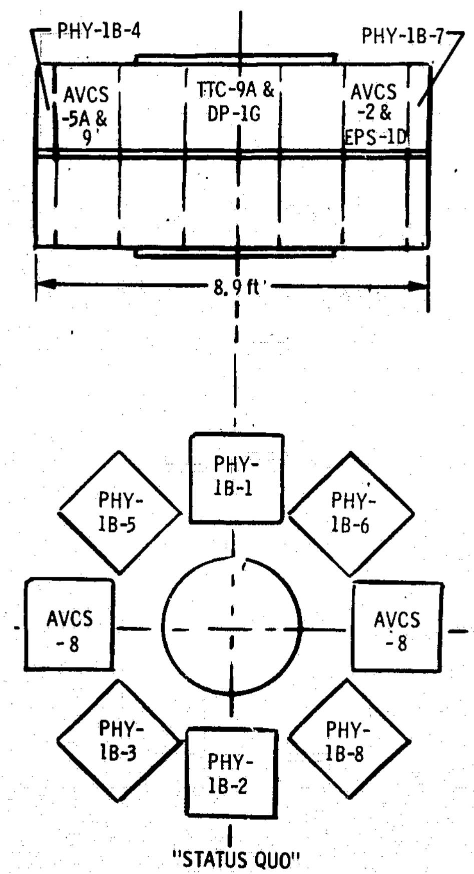
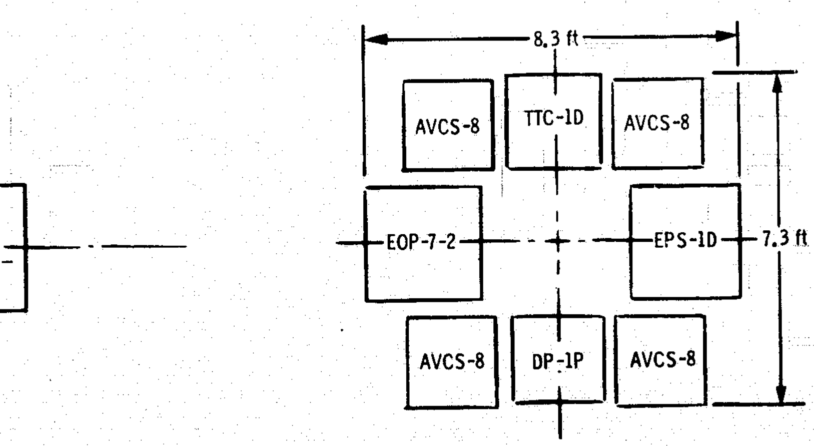


Figure V-5 Upper Atmosphere



ty Satellite

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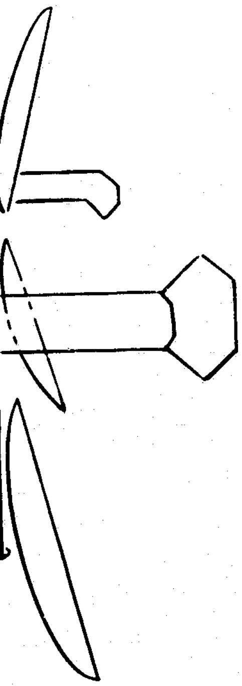


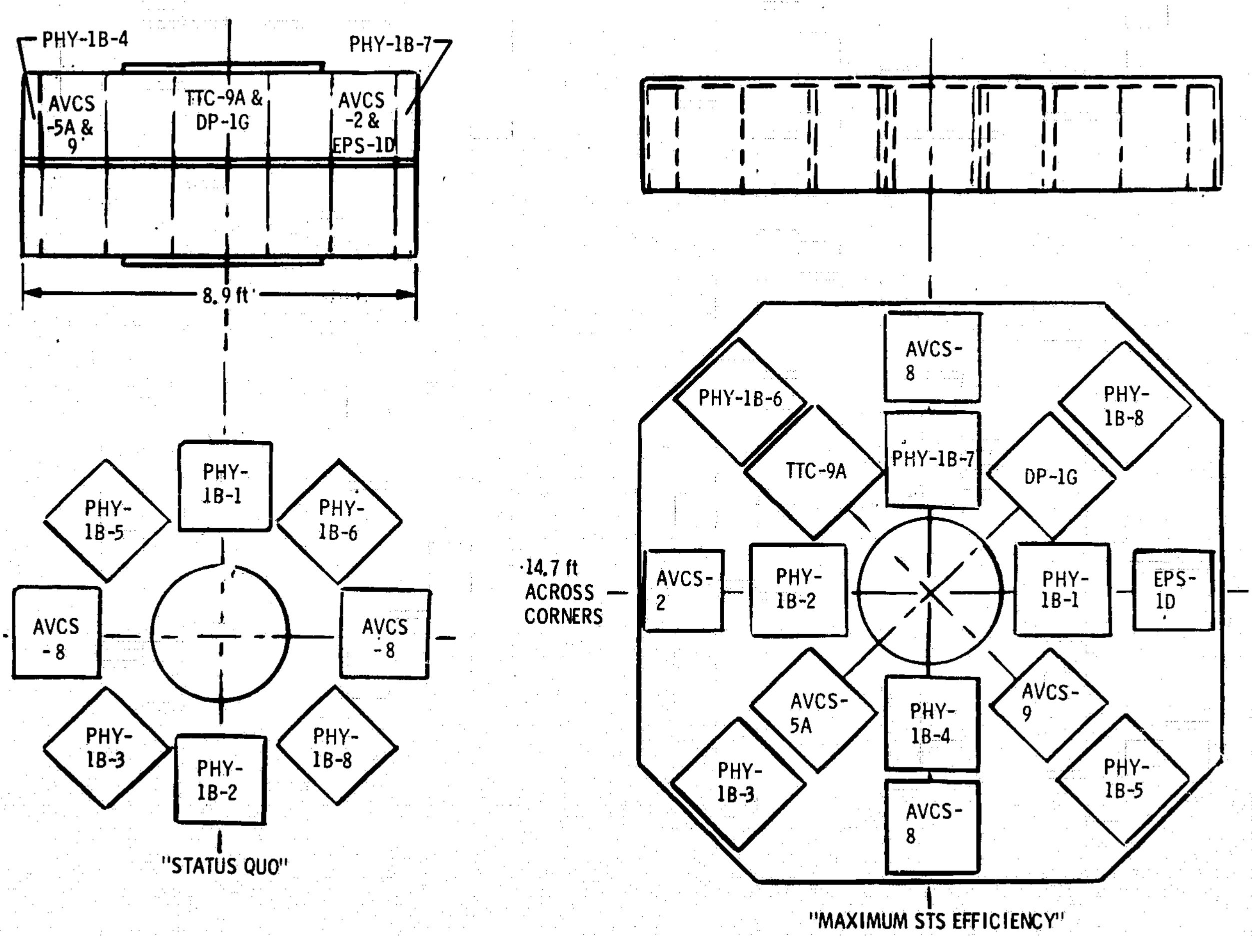


Figure V-5 Upper Atmosphere Explorer

STS efficiency" form of the UAE takes advantage of the full orbiter cargo bay diameter and is commensurate with the tug diameter. Should the launch cost reimbursement policy include a cost increment based on cargo bay length, then the flat disk form of spacecraft will be more widely used. The "maximum STS efficiency" form of the UAE requires one docking interface for the pivoting arm servicer and cannot be easily serviced by the general purpose manipulator.

The flat disk configurations; e.g., Fig. V-1, suit axial module removal by the pivoting arm. When the general purpose manipulator is used for radial module removal then the modules toward the center cannot be reached directly. The alternatives include lining the module up two deep as in right-hand side of Fig. V-5 and taking out the outboard module and then the second module. Another approach is to use two tiers of modules as shown on the left-hand side of Fig. V-5. Generally, radial module removal results in some loss of access to the central area of the spacecraft. This loss in volumetric efficiency can be up to 30%. The axial removal approach suffers when the "status quo" configuration is used and the spacecraft takes the form of two tiers of modules. In this case, the pivoting arm must be docked separately at each end of the spacecraft. The spacecraft configurations used in the reference documentation were also reviewed. It was found that all of the specific spacecraft configurations were adaptable to axial module removal and some used radial module removal. Thus, axial module removal has a slight preference.

The general purpose manipulator can perform axial module replacement by using a central docking system that provides adequate module removal space between the spacecraft and the stowage rack. Similarly, the pivoting arm can be extended and degrees-of-freedom added so that it can reach outside the spacecraft and replace modules radially.

In addition to the mechanical fastening between SRUs and the spacecraft, the following possible connector needs were identified 1) electrical, 2) waveguide, 3) fluid, and 4) thermal. Black box electrical connections in spacecraft have tended to run to the hundreds of pins spread over several connectors per box. The problems of mating and demating this large number of pins seems difficult. The weight of electrical harness involved has been leading towards signal multiplexing or the data bus approach. This results

in a few signal and data pins per box and effectively provides many more inputs and outputs through the multiplexing function. A similar trend exists with regard to electrical power. Power is distributed at one voltage to all boxes, then each box converts to the variety of voltages and frequencies needed within the box. In this way the total number of electrical connections per box, including redundancy, can be kept to less than 20 in one or two connectors. This level of connectors and pins is very adaptable to integration into the mechanical fasteners.

Waveguide connector adaptations to serviceable spacecraft have been addressed by COMSAT Labs in their parallel study. They have conceived a basis for waveguide connections for space serviceable spacecraft. It has been possible to avoid fluid connections in our spacecraft configuration analysis and they were avoided in all of the prior studies. In each case a thruster set and its propellant tank were packaged in the same module, with four modules per spacecraft to provide full attitude control. This indicates that fluid connectors are not necessary. However, there are some advantages in being able to interconnect propellant tanks to compensate for some kinds of RCS failures. It is thus recommended that development of fluid connectors be initiated.

The thermal aspects of serviceable spacecraft have been addressed to a first level with most approaches being to treat each module separately and minimize inter-module heat flow. The PUT study (Chapter XI, Item K-8) did use heat pipes to conduct heat from the inner modules to the spacecraft exterior. The UOP study (Chapter XI, Item M-13) incorporated heat flow paths through its baseplate to conduct heat from modules to the far side of the spacecraft. This suggests that the development of a thermal conductor be initiated to increase the number of thermal design alternatives available.

The question of module alignment accuracy was briefly addressed. Based on Martin Marietta's LST study contract information, we concluded that instrument change-out is feasible in most cases. Alignment accuracy requirements will in some cases require the development and addition of different calibration techniques. The current SRU interface mechanism designs are predicted to have an alignment accuracy of 0.001 in. in translation and 15 arc seconds in rotation. This accuracy is a factor of four better than the accuracy

required for the replacement of the area photometer in the LST. Both Itek and Perkin-Elmer have developed alignment guide designs for LST which provide alignment of the instruments and guidance sensors when they are replaced as modules by EVA. Their design accuracies are: 0.004 in. in translation and 50 arc seconds in rotation. This data has led to the conclusion that instruments/sensors can be replaced with a servicer mechanism in most cases without a refinement to the current interface mechanism design. It is felt that a factor of five improvement could be realized if the alignment pin design was refined. Testing should be done on latches under realistic environmental conditions to verify these numbers.

Even with the alignment accuracies as quoted, there will certainly be some instruments which require special hardware development in the area of mounting and in-space calibrating. An obvious desirable guideline is to locate hardware requiring critical relative alignment in the same module. Thus, the alignment can be performed on the ground. An example is a set of attitude reference gyros. Their relative alignment is critical and can be done on the ground. However, their alignment as a unit in the spacecraft is not critical unless they are involved in a navigation process.

Developing special calibration techniques needs to be explored. It appears that by programming the spacecraft through certain maneuvers after servicing, calibration data could be obtained from the sensors and instruments. Alignment errors could then be calculated with a computer program. The use of known alignment errors would then depend on the specific mission.

The results of this section may be summarized as:

1) To a first level, spacecraft can be made serviceable without mission objective penalties -- more a management challenge to coordinate development,

2) Further iterations are necessary to identify proper packaging density/weight penalties for servicing,

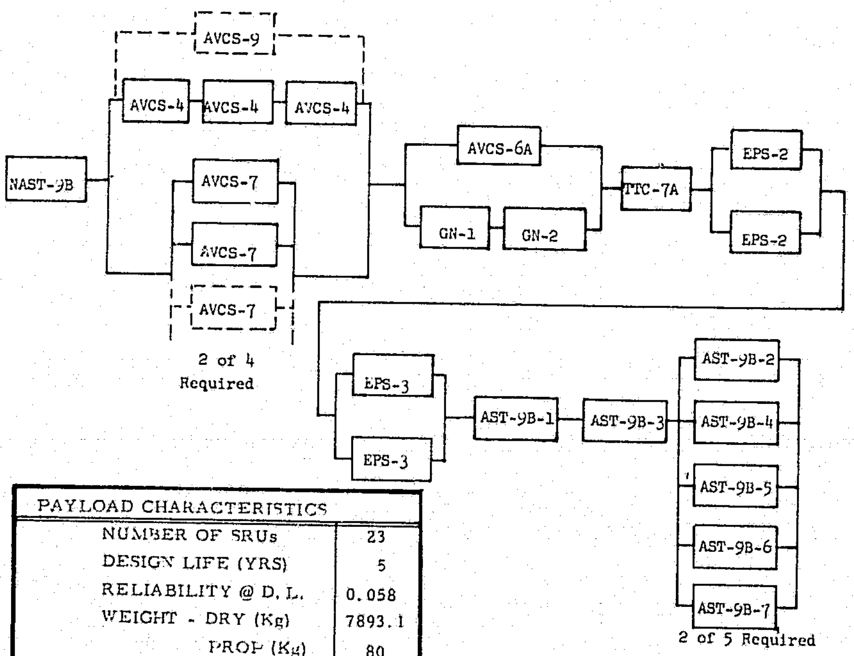
3) Configuration along the policy lines of "status quo" or "maximum STS efficiency" has more impact to spacecraft than servicer, and

4) Servicers selected appear to accommodate a wide range of spacecraft characteristics without excessive configuration penalties.

The above analysis used standardized subsystem modules from the OA study. This indicates that on-orbit servicing in the form of module exchange is compatible with standardized subsystems. However, there was nothing in the above analysis that required that the modules be standardized. The SRU interface mechanisms must be standardized, but the modules need not be standardized.

B. SPACE-REPLACEABLE UNIT CHARACTERISTICS

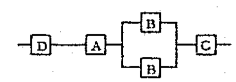
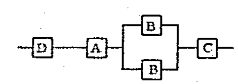
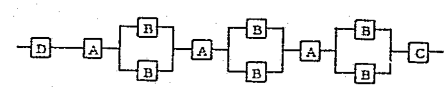
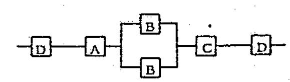
The characteristics of the space-replaceable units, or modules, that were used have been drawn primarily from the Operations Analysis study (Chapter XI, Items F-14 through 27). The module data is briefly described here for reference purposes. The OA study considered 29 spacecraft of which 26 were in the IOSS maintenance applicable set. They examined the SSPD definition of each spacecraft, each spacecraft subsystem, and the major subsystem elements. They cross-compared the data for each subsystem and arrived at a set of standardized subsystem modules. Table V-1, taken from Item F-20 of Chapter XI, shows the form of the data for the attitude and velocity control subsystem. The spacecraft were then configured from these standard modules as shown in Fig. V-6 for the Large X-Ray Telescope, also from Item F-20 of



PAYLOAD CHARACTERISTICS	
NUMBER OF SRUs	23
DESIGN LIFE (YRS)	5
RELIABILITY @ D. L.	0.058
WEIGHT - DRY (Kg)	7893.1
PROP (Kg)	80
TOTAL (Kg)	7973.1

Figure V-6 Space Serviceable Large X-Ray Telescope

Table V-1 Attitude and Velocity Control Standard Modules

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10[illegible]/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
AVCS-1	Reaction Wheel (5 ft-lb-sec)	A	Reaction Wheel	1	4.5	4.5	700	10	0.718	26.16	1.224	
		B	Wheel Electronics	2	1.4	2.8	6000					
		C	Remote Terminal	1	2.0	2.0	500					
		D	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			40.3						
AVCS-2	Reaction Wheel (10 ft-lb-sec)	A	Reaction Wheel	1	8.2	8.2	700	10	0.718	26.16	1.224	
		B	Wheel Electronics	2	1.4	2.8	6000					
		C	Remote Terminal	1	2.0	2.0	500					
		D	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			44.0						
AVCS-3	Reaction Wheel ([illegible] lb-sec/ wheel)	A	Reaction Wheel	3	24.5	73.5	700	7	0.613	13.18	1.270	
		B	Wheel Electronics	6	2.7	16.2	6000					
		C	Remote Terminal	1	2.0	2.0	500					
		D	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			122.7						
AVCS-4	Control Moment Gyro (Double Gimbal) (600 ft-lb-sec)	A	CMG Wheel	1	68.0	68.0	700	2	0.944	30.24	1.063	
		B	Wheel Electronics *	2	4.5	9.0	6000					
		C	Torquer, Damper and Resolver	2	4.5	4.0	1000					
		D	Remote Terminal	1	2.0	2.0	500					
		F	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			119.0						

Chapter XI. The figure also lists some pertinent summary data as well as the mission equipment required. The details of the LXRT mission equipment are given in Table V-2 from Item F-19 of Chapter XI. The mission equipment was taken directly from the SSPD and then the serviceability aspects of base-plates, environmental control, mechanism, and electrical distribution and power conditioning were added by Aerospace Corporation.

Spacecraft module packaging density is one of the drivers affecting both spacecraft configuration for servicing and the identification of performance overhead due to the module stowage rack and STS interfaces of both the tug and orbiter. Shown in Table V-3 is a brief sample of the range of packaging densities found in the literature along with the values used in this IOSS. These densities determine the volume required, center of gravity migration, ACS stabilization propellant, and spacecraft to servicer interface geometry. Therefore, since the range of values is relatively great, subsequent studies should identify and refine this impact.

Table V-3 Module Density Comparison

	DENSITY, lb/ft^3			
	COMSAT NONSERVICEABLE	TRW SSC-3 SERVICEABLE	AEROSPACE EOS SERVICEABLE	IOSS
POWER-BATTERIES, CHARGER, MONITOR, ETC	87	25	35.4	35
ATTITUDE CONTROL-PROPELLANT AND EQUIPMENT	NOT AVAILABLE; SHOULD BE SOME DUE TO PROPEL. WEIGHT	21	19.6	22
ELECTRONIC EQUIPMENT-COMM., GUIDANCE, TELEMETRY AND TRACKING	33.4	9.5	11.5	15
REACTION CONTROL WHEELS	107	NOT UTILIZED	30	30
NOTE: THE 3:1 REDUCTION IN DENSITY WHEN MODULARIZATION IS USED.				

Another basis for estimating module sizes is the data pertinent to the on-orbit servicer stowage racks. This data is shown in Fig. V-7 for eight

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Table V-2 Mission Equipment Modules

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
AST-9B	Focusing X-ray Telescope - 3.0M	AST-9B-1		Field Monitor Camera & Elect.	1		23.0	1	1	2	.990	199.00	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			77.8						
		AST-9B-2		Guide Star Trackers & Elec.	2		18.0	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			72.8						
		AST-9B-3		Position Sensing Prop. Counter & Elec.	1		34.0	1	1	2	.990	199.00	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
		AST-9B-4		X-ray Image Det./Int. & Elec.	1		59.0	3	4	2	.560	3.45	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			113.8						
		AST-9B-5		Crystal Spectrometer & Elec.	1		43.5	2	3	2	.680	5.19	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			98.3						
		AST-9B-6		Max. Sen. Det. & Cryo. & Elec.	1		41.0	3	4	1	.560	1.72	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			95.8						
		AST-9B-7		Lithium Hydride Polar. & Elec.	1		40.0	1	4	2	.840	11.47	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Control	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			94.8						

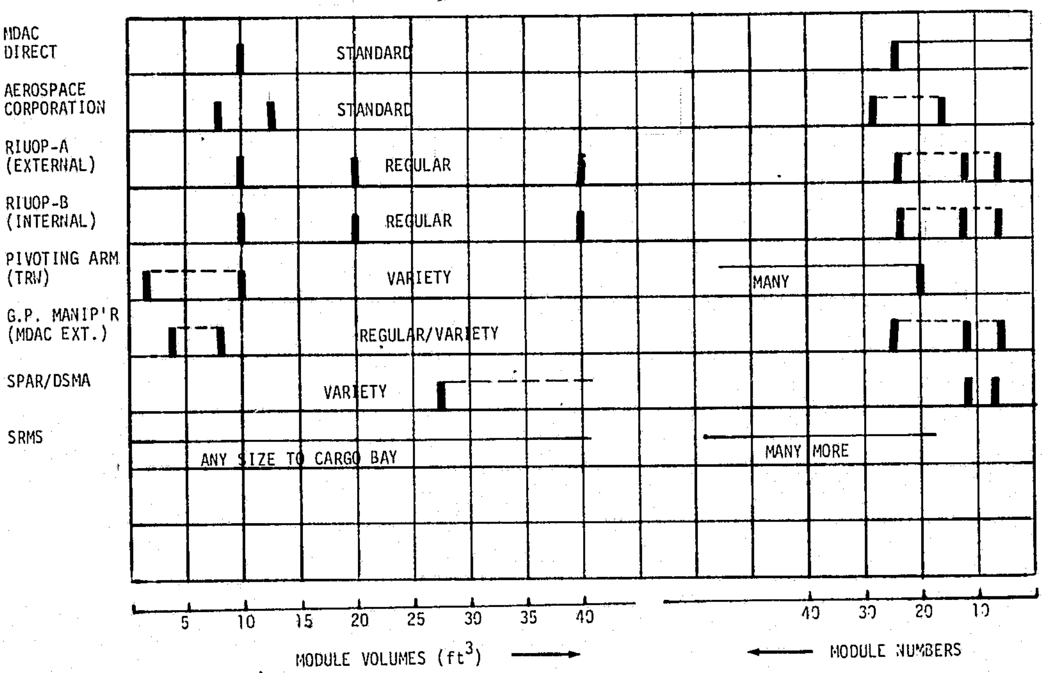


Figure V-7 Module Characteristics of Servicer Stowage Racks

different on-orbit servicers. The module volumes range from two to 40 cubic feet, corresponding to 1.25 to 3.4 foot cubes, and then up to the size of the orbiter cargo bay for the SRMS. The number of modules that can be stowed in the stowage rack are generally eight to 28 with the exception of the pivoting arm and the SRMS. The DSCS II study examined three levels of modularization for the DSCS II with 8, 15, and 30 modules. The lightest module was 7 lbs and the heaviest 487 lbs with corresponding volumes of 1.8 ft^3 and 32 ft^3. Average nonpropulsive module weights were 34, 68, and 181 lbs for the three SRU sizes. The DSCS II propulsion modules were large; 200 and 450 lbs for reaction control and stationkeeping, respectively, because of the DSCS II ground rule to initially provide enough on-board propellant for a ten-year lifetime. The EOS studies used just four subsystem modules, each of which was 24 ft^3 with weights ranging from 100 to 300 lbs. They also used three pieces of mission equipment weighing from 60 to 360 lbs. It can thus be seen that the range of module sizes and weights considered have covered a wide range. However, the OA study weights and the Martin Marietta sizes (section A above) can be taken as representative of what can be done if an attempt is made to avoid the outsize modules. The result is a set of

spacecraft configured from reasonably sized modules that can be effectively handled by an on-orbit servicer mechanism.

C. SPACE REPLACEABLE UNIT INTERFACE MECHANISMS

The maintenance concept analysis of Chapter IV and the spacecraft configuration analysis above led to the conclusion that the SRU interface mechanisms represented a very significant interface and thus required further study. An effort was initiated to identify the criteria or design guidelines for an SRU interface mechanism and then to search the literature for a suitable concept. The process identified some 12 interface mechanisms. None of the concepts satisfied all of the design guidelines, although the Grumman Aerospace Corporation earth observatory satellite concept was better than any of the others. This resulted in an interface mechanism based on the Grumman and TRW DSCS II concepts being designed. This concept involved side-mounting of modules in the spacecraft. After discussion with MSFC of the relative merits of side-mounting modules versus bottom-, or end-mounting of modules and the implications of these two approaches on spacecraft structure, it was decided that a bottom-mounted SRU interface mechanism, based on an MSFC concept, would be designed and that engineering test units of both the bottom- and side-mounted units would be built along with a single end effector adaptable to both mechanisms.

Of the various interfaces between spacecraft and servicer and spacecraft and rendezvous and docking system, it became apparent that the SRU interface mechanisms were quite significant. The servicer mechanism must attach to each module so it can transport the module between the stowage rack and the spacecraft. Each module must be latched into the spacecraft and into the stowage rack. Associated with these operations are mating/demating of connectors of several types and status indication. As there is one interface mechanism for each module, the cumulative weight and volume effect is important. There also is an important interaction between the interface mechanism "capture volume", the servicer mechanism accuracy, and the servicer control system.

1. Interface Mechanism Requirements

The elements involved in the attachment of the servicer mechanism end effector to the module (designated the attach) and the module to the spacecraft or stowage rack (designated the latch) are shown in Fig. V-8. The

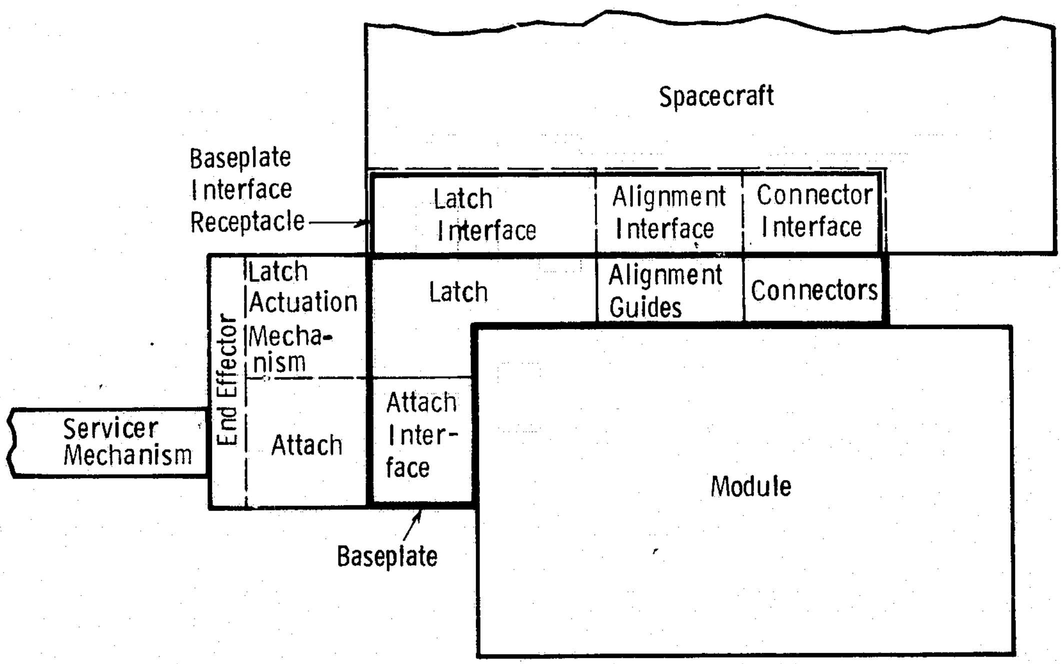


Figure V-8 Interface Mechanism Elements

elements shown represent the parts of the system involved. It is important that all the elements be considered when setting down design guidelines and evaluating existing designs. A range of terminology has been used by industry in the module exchange arena. This results naturally in some difficulty in understanding and comparing designs. The subassemblies and elements shown in Fig. V-8 are an attempt to establish an explicit terminology for this study. As seen in the figure, the attachment mechanism used to attach the end-effector to the module is called the attach. The latching mechanism used to latch the module to the spacecraft or stowage rack is called the latch. The interfaces bewteen the end effector, module, and spacecraft are performed by what is designated as a baseplate which hard-mounts to the module and a baseplate interface receptacle which hard-mounts to the spacecraft. A similar baseplate interface receptacle is used between the baseplate and the stowage rack.

The functional steps necessary for module exchange were identified and broken down to a sufficient level of detail to allow the identification of alignment interfaces and error sources to be performed correctly. The

alignment was considered in six degrees of freedom for each subassembly and element, and the location of the coordinate system was also considered. The alignment interfaces for each step of the module exchange sequence were identified so that the error contributions could be properly assigned. Some of the error sources contributing to the misalignments which must be accommodated to realize the desired final alignment are given in Table V-4. The determination of the magnitudes of these errors and their contributions to the

Table V-4 Alignment Error Sources

SERVICER MECHANISM
END EFFECTOR (ATTACH/LATCH ACTUATION)
BASEPLATE
ATTACH INTERFACE
LATCH
ALIGNMENT GUIDES
CONNECTORS
BASEPLATE INTERFACE RECEPTACLE
LATCH INTERFACE
ALIGNMENT INTERFACE
CONNECTOR INTERFACE
SERVO POSITIONING ACCURACY
DOCKING SYSTEM ACCURACY
LOCATION OF LATCH ACTUATOR AXIS RELATIVE TO ATTACH AXIS

misalignments of the attach and latch interfaces must await a more detailed definition of the docking system as well as initiation of a subsequent study.

The selected system and element level guidelines are given in Table V-5. Many of these are logical extensions of the servicer mechanism criteria and need not be discussed further. By requiring the attach and latch forces and moments to be transmitted through the interface mechanism structure (baseplate), the spacecraft module designer does not have to provide spacecraft load-carrying paths within the module. Also, the load-carrying paths can usually be kept shorter by controlling them within the baseplate and its mating receptacle. This should result in a weight savings. The servicer mechanism should only be designed to transmit low level forces for initial capture and alignment but not final lockup. A significant savings in weight results from this approach. When the attach-latch forces and moments are applied within the end effector and baseplate, a balanced load path geometry can be realized and the loads can be contained within the immediate structure. Simulations at Martin Marietta have indicated that a force of 20 lbs should be adequate for module exchange.

Table V-5 Design Guidelines

<u>ATTACH-LATCH SYSTEM LEVEL DESIGN GUIDELINES</u>
Impose Minimum Restrictions on the Spacecraft and Module Designers - Allow Flexible and Efficient Packaging of Modules on Spacecraft and Stowage Device - Accommodate a Wide Range of Module Sizes and Masses - Baseplate Transmits All Forces and Moments - Accommodate a Range of Connector Types and Forces
Accommodate Misalignment in Six Degrees of Freedom
Minimize Weight and Volume
Require Servicer Mechanism Forces of Less Than 20 lbs
Be Compatible with Operation by Astronaut
Provide Nonredundant Module Support
Accommodate Orbiter Crash Loads
Allow for Thermal and Structural Deflections
<u>ATTACH-LATCH ELEMENT DESIGN GUIDELINES</u>
<u>APPLICABLE TO ATTACH AND LATCH</u>
Provide a Two-Stage Engagement: Capture and Lockup
Provide Separation Forces
Generate Operational Status Signals
Utilize an Actuator Located in End Effector
Accomplish Capture under Required Misalignment Tolerances (6 DOF)
Make Final Alignment to Required Accuracy (6 DOF)
Minimize Sliding Friction Areas
<u>ATTACH ONLY</u>
Use a Passive Interface on Baseplate
<u>LATCH ONLY</u>
Use a Passive Interface on Spacecraft
Provide Load Paths at Final Alignment to Handle Orbiter Crash Loads
Avoid Initial Module to Opening Close-Fit Requirement
Provide Positive Lockup Device
Provide Connector Make/Break Forces

The non-redundant module support approach is recommended to minimize module location inaccuracies and to avoid transferring bending and thermal loads to or from the spacecraft and module. A two-stage engagement, capture and lockup, is dictated by the fact that capture misalignments are considerably larger than the final alignment requirements. The two stages also result from the fact that the forces applied during the capture stage come from the servicer mechanism and during the lockup stage they come from an actuator in the end effector. The recommended approach of using a latch mechanism

actuator located in the servicer mechanism end effector results in (1) a single actuator for each servicer, (2) simple control via electrical signals, (3) a system that can be made very stiff, and (4) a lower overall system weight. The disadvantage is the need for a mechanical drive-power interface, an interface which is not very difficult.

One approach that seems useful, though not categorized as a design guideline, is to think of the interface mechanism as being made as a two-part kit - one part of the kit to be mounted on the spacecraft and the other part to be mounted on the module. In this way each designer, spacecraft and module, would know what the interface was and thus would not need to develop a new latch concept for each spacecraft. Rather, one kit concept might be used across all spacecraft. The variety of module sizes might dictate several sizes of latch kits, but they could all use the same basic design and the same attach interface.

2. Interface Mechanism Evaluations

This paragraph reflects the processes by which the 12 interface mechanism concepts of the literature were evaluated against a set of criteria applicable to the wide range of spacecraft and modules contained in this study's maintenance applicable set. The 12 concepts were associated with the 15 servicer concepts discussed elsewhere in the study, augmented by three recent design studies on the earth observatory satellite (EOS). None of the interface mechanism concepts satisfied all the criteria or design guidelines because each of the concepts was designed to suit a limited spacecraft sample or was constrained by a particular servicer mechanism. The Grumman Aerospace Corporation EOS concept is better than the other concepts with the TRW DSCS II concept also satisfying a large number of the criteria.

It was not intended to develop a relative ranking of interface mechanism concepts; rather the intent was to identify, from the literature, an existing concept that could be further defined. While the search was unsuccessful, much was learned that helped lead to our design.

The 12 latch/attach mechanisms identified are listed in Table V-6 by reference to the originating organization with modifiers as necessary. The mechanisms have been grouped, in the case of the EOS designs, where four functionally similar alternatives were created by four different organizations.

Table V-6 SRU Interface Mechanisms

McDONNELL DOUGLAS ASTRIONICS COMPANY DIRECT ACCESS (K-8)
AEROSPACE CORPORATION (F-19)
BELL AEROSPACE CARTESIAN COORDINATE (K-1)
ROCKWELL INTERNATIONAL - UOP-B (INTERNAL) (M-13)
ROCKWELL INTERNATIONAL - GEOSYNCHRONOUS PLATFORM DEFINITION STUDY (M-6)
TRW SYSTEMS INC. PIVOTING ARM (N-14)
MSFC - PUSHROD LINKAGE CONCEPT } INFORMAL
MSFC - CABLE LINKAGE CONCEPT } COMMUNICATION
SPAR/DSMA CARGO BAY ONLY - EOS (O-1)
GRUMMAN AEROSPACE CORPORATION - EOS (I-8)
GENERAL ELECTRIC - EOS (H-12)
TRW SYSTEMS INC. - EOS (N-7)

The primary specific references used for the detail data on the concepts of the table are given in Chapter XI by the item numbers in parentheses.

The important ground rules used in this evaluation were: (1) spacecraft designed to be serviceable, (2) module exchange only, and (3) all modules are located on one or two separate docking faces or in one or two adjacent tiers. The various detail mechanisms identified in the interface mechanisms from the literature were related to the system and element guidelines developed above to obtain an idea of which mechanisms might be more useful and which should not be used for the interface mechanism application. Twelve mechanisms such as push-off devices and ball-screw actuators were identified. The full list of system and element level guidelines was reviewed and 11, such as baseplate transmits all forces and moments and minimize sliding friction areas, were selected for use in this part of the evaluation. The result of the evaluation is that seven mechanisms were judged to be good, simple, highly reliable items that will be dependable when properly utilized in a design. These are: (1) push-off devices, (2) cone/wedge locators, (3) worm gear drives, (4) linkage systems, (5) hook latches, (6) rollers, and (7) ball screw actuators.

The 12 interface mechanism concepts of Table V-6 were evaluated against a preliminary set of the system and element level guidelines of Table V-5 as part of the mechanism screening. Table V-7 is a summary of the evaluation. Each interface mechanism design was analyzed to determine if it could functionally

Table V-7 Summary of Latch/Attach Evaluation Against Design Guidelines

	MDAC Direct Access	Aero-space Corp.	Bell Aero-space	RI UOP B	RI Geosynch. Platform	TRW DSCS II	MSFC Link-ages (latch only)	MSFC Cables (latch only)	SPAR/ DSMA EOS	Grumman EOS (latch only)	G.E. EOS	TRW EOS
ATTACH-LATCH SYSTEM LEVEL DESIGN GUIDELINES												
• Impose Minimum Restrictions on the Spacecraft Designer												
– Accommodate a Wide Range of Module Sizes and Masses	No	Yes	No	Yes	No	Yes	No	Yes	No	Mod	No	No
– Baseplate Transmits All Forces and Moments	No	Yes	Yes	Yes	Yes	Yes	Yes	No	Yes	Yes	Yes	Yes
– Accommodate a Range of Connector Types and Forces	Mod	No	Mod	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes
• Accommodate Misalignment in Six Degrees of Freedom	No	No	No	No	No	Yes	Yes	Yes	No	Yes	No	Yes
• Minimize Weight and Volume	No	No	Yes	No	Yes	Yes	No	Mod	No	Mod	No	No
• Require Servicer Mechanism Forces of Less than 10 lbs	No	Mod	Yes	No	No	Yes	Yes	Yes	No	Yes	No	Yes
• Be Compatible with Operation by Astronaut	No	Mod	Yes	No	No	Yes	Yes	Yes	No	Yes	No	Yes
• Provide Nonredundant Module Support	No	No	No	No	No	Yes	Yes	Yes	No	Yes	No	No
• Accommodate Orbiter Crash Loads	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod
ATTACH-LATCH ELEMENT DESIGN GUIDELINES												
APPLICABLE TO ATTACH AND LATCH												
• Provide a Two-Stage Engagement: Capture and Lockup	No	Yes	No	No	No	Yes	Yes	Yes	No	Yes	No	Yes
• Provide Separation Forces	Yes	Yes	Yes	No	No	Yes	Yes	Yes	Yes	Yes	Yes	Yes
• Generate Operational Status Signals	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod
• Utilize an Actuator Located in End Effector	No	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes
• Accomplish Capture Under Required Misalignment Tolerances (6 DOF)	Yes	Yes	No	No	No	Yes	Yes	Yes	No	Yes	No	Yes
• Make Final Alignment to Required Accuracy (6 DOF)	No	No	No	No	No	Yes	Yes	Yes	No	Yes	No	Yes
• Minimize Sliding Friction Areas	No	Yes	No	No	No	No	Yes	Yes	No	Yes	No	No
ATTACH ONLY												
• Use a Passive Interface on Baseplate	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes
LATCH ONLY												
• Use a Passive Interface on Spacecraft	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes
• Provide Load Paths at Final Alignment to Handle Orbiter Crash Loads	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod	Mod
• Avoid Initial Module to Opening Close-Fit Requirement	No	Yes	No	Yes	Mod	Yes	Yes	Yes	No	Yes	Yes	Yes
• Provide Positive Lockup Device	Yes	No	No	Yes	Yes	No	Yes	Yes	No	No	No	No

satisfy each of the design guidelines. Thus a design had to have a functional capability but not necessarily an optimum design mechanization. If the design did not satisfy a guideline, it was given a "no". If it was felt that a minor modification would allow the design to then satisfy a guideline, it was designated by "Mod" in the table.

None of the interface mechanism concepts satisfied all the system and element level design guidelines. This is mainly due to the fact that each concept was designed to satisfy module exchange for a limited spacecraft sample or was constrained by a particular servicer mechanism. The Grumman Aerospace Corporation EOS interface mechanism concept satisfied from a functional capability standpoint all but one of the guidelines. The TRW DSCS II and the MSFC linkage and cable designs were next in satisfying the largest number of the guidelines.

A mechanical design alternatives evaluation was used to examine each of the 12 interface mechanism designs against the 12 mechanisms found in the various designs. The evaluation was not to pick a best design, but rather to identify that design(s) which satisfies all the requirements. Each design was reviewed to identify which mechanisms it used. The Grumman EOS design employed none of the bad mechanisms, all four of the good mechanisms, and two out of the three acceptable mechanisms. Thus, it incorporates a good set of mechanisms overall and is satisfactory for this evaluation. The two MSFC concepts used four mechanisms from the good and acceptable categories, with only the cables falling in the bad category. The TRW-DSCS II design was interesting in that it incorporated only two good and one bad mechanisms and thus is relatively simple. However, if the single screw is deleted, it could be replaced by a combination of a worm gear drive, linkage system, hook latches, and rollers. The result would be a more complex design using all the good and two of three acceptable mechanisms in what could be a more compact representation of the Grumman EOS design.

The observations and recommendations reached from this analysis and evaluation of interface mechanisms are:

1) None of the 12 attach/latch designs uncovered in the literature completely satisfy the design guidelines used,
2) The Grumman EOS design comes closest to meeting the design guidelines and uses the better mechanisms,

3) The design guidelines postulated and used in this analysis appear valid and may be used as the basis of further work, and

4) A combination of the Grumman EOS and the TRW DSCS II designs might well be used as the basis of an effective interface mechanism.

3. Spacecraft Structural Considerations

While the interface mechanism evaluation did not explicitly address spacecraft structural arrangements, they are an implicit part of the considerations. Each of the servicer mechanisms as well as the interface mechanism designs was developed with a spacecraft structural concept in mind. One expression of this structure is in terms of how the modules geometrically relate to the structure. The Bell Aerospace approach has the modules inset in the spacecraft, like drawers. The RI-UOP-B approach has the modules fastened to trays which are "bottom" mounted to a flat spacecraft surface. The TRW DSCS II approach has the modules "side" mounted to a deep web structure. The EOS approach is to "corner mount" modules to an open frame structure. The Martin Marietta preference, from a mechanisms point-of-view, is for the TRW DSCS II approach. The deep webs carry loads very efficiently in shear. The basic load carrying structure can be less deep than the modules, thus the structure does not add to the spacecraft depth as does the UOP floor. The modules can be mounted outside the webs and thus the spacecraft structure can be smaller, and hopefully lighter. The electrical cables can be run flat along the shear webs and interconnected in the corner thus using the space efficiently. A web is soft in torsion about a line in the plane of the web, but this can be overcome by using two webs in parallel, properly spaced and interconnected by other webs. The webs can be locally reinforced by hat sections or honeycomb to pick up concentrated loads.

When this deep-web approach is used, the modules are "side" mounted to a web. This has led, partially, to a preference for side mounting modules as compared to bottom or corner mounting of modules. This preference was reflected in the interface mechanism evaluations. The deep-web form of spacecraft structure puts the structure inside where its thermal control is easier, and the heat sources in the modules outside where they can radiate to deep space. The side mounting of modules makes a TV system more useful in that it can be mounted on the mechanism end effector in a way so that it can see both the

attach interface and the latch interface. In this way it can be used to control (or confirm) attachment and latch operation.

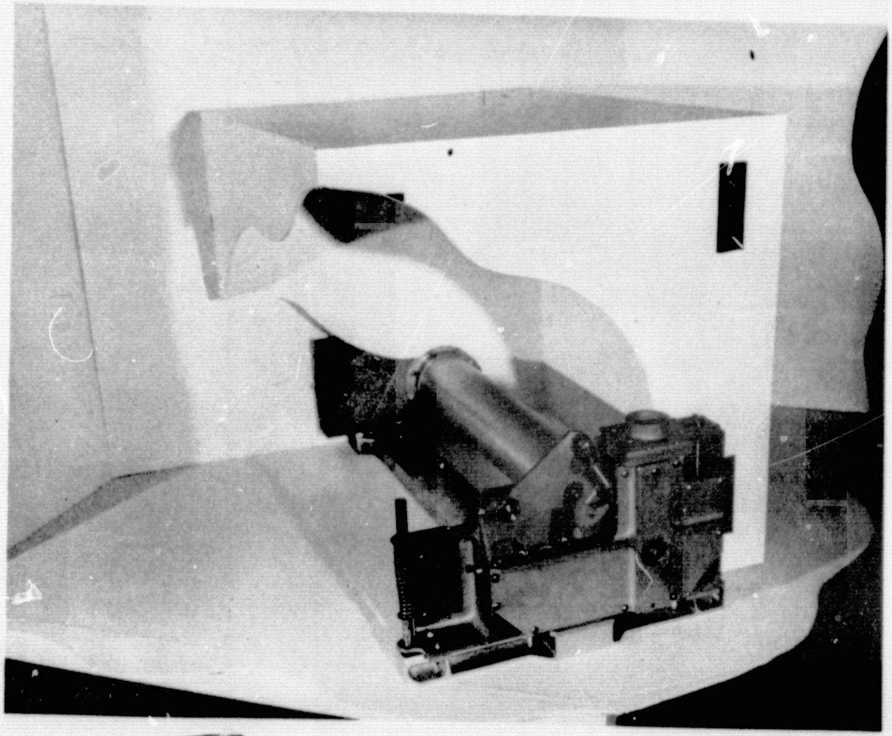
The MSFC interface mechanism using the bottom-mounting approach, has been evolving for some time as an outgrowth of the unmanned orbital platform definition study that MSFC has been using to obtain a better understanding of standardized spacecraft. These standardized spacecraft were designed so that all the SRUs were mounted to a single flat face. This provides a broad latitude in selecting location and orientation of each module and the area is not broken up by a series of webs so that the area may be efficiently used. The modules are all effectively mounted outside so the modules can radiate to deep space.

There was significant discussion as to which of the side- or bottom-mounting SRU interface mechanisms is more applicable to the design of serviceable spacecraft. No answer was found. Both concepts have value. It was thus suggested that it might be advantageous to design and fabricate engineering test units of each latch mechanism and of a compatible on-orbit servicer end effector. The two approaches, bottom and side, need not be thought of as competing approaches, but rather both are viable alternatives that will give designers of serviceable spacecraft more options in their selection of spacecraft configurations.

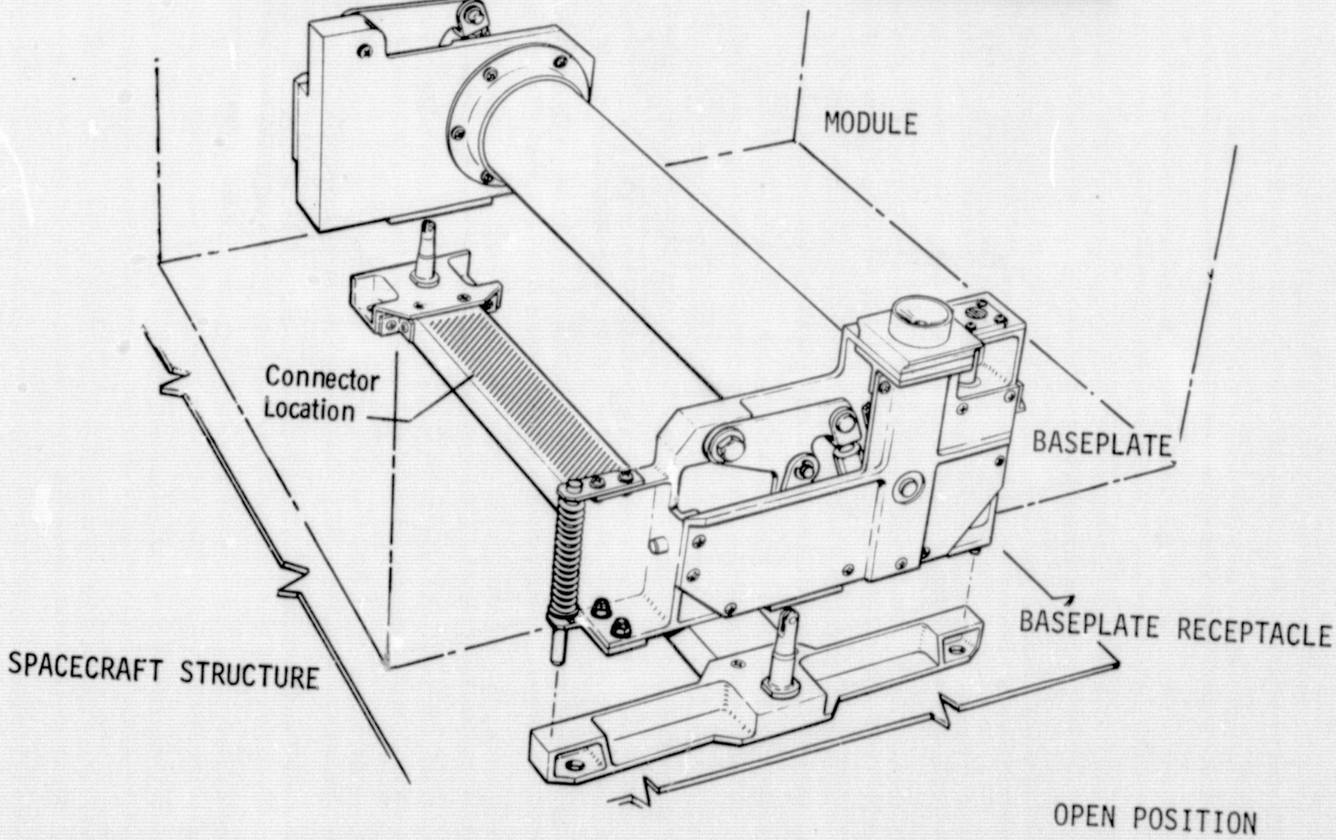
4. Bottom-Mounted Interface Mechanism

A bottom-mounted SRU interface mechanism was designed. It was based on the MSFC designs and meets all the system and element level guidelines using those mechanisms that have previously been defined as good or acceptable. This system will accept SRU modules over a range of sizes from a 15-inch cube to a 40-inch cube. The engineering test unit that has been fabricated will accept a 26-inch module.

The design is shown in Figure V-9. The SRU module mounts on the baseplate and the baseplate receptacle mounts to the spacecraft. Connector locations are between the attach pins on the baseplate receptacle. Corresponding locations would be on the baseplate. The baseplate mechanism details are shown in Figure V-10. The baseplate is installed over the pins on the baseplate receptacle. Initial alignment is provided by the cones on the bottom of the



CLOSED POSITION



OPEN POSITION

Figure V-9 Bottom-Mounting Interface Mechanism

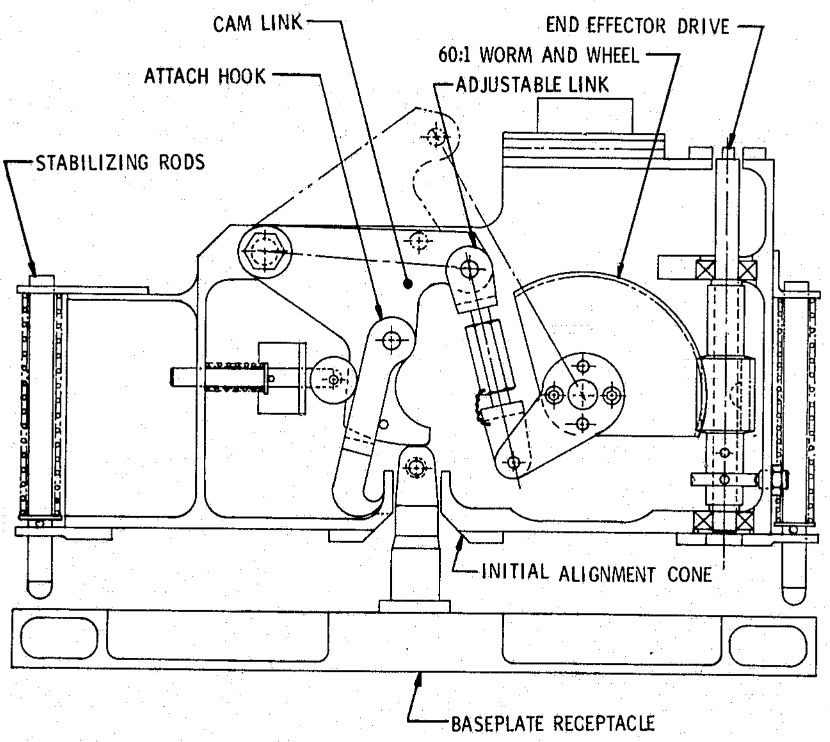


Figure V-10 Baseplate Mechanism Details - Bottom Mounting

baseplate. Outrigger spring-loaded rods give the module stability and prevent binding on the receptacle pins. After initial alignment is obtained, the 60:1 worm gear is rotated. This activates the linkage, and hooks on both sides of the module engage the rollers in the receptacle pins. Torque is transmitted to the linkage on the opposite side of the module via a torque tube. The module is between 1-1/4 and 1-1/2 inches above final lockup and initial hook engagement. As the baseplate and receptacle are drawn closer together, progressively larger diameters on the receptacle pins improve alignment until final alignment is obtained approximately 0.6 inch above final lockup.

The mechanism provides approximately 300 pounds of force at initial hook engagement. This force increases as the parts come closer together due to the toggle action of the mechanism. To maintain lockup the toggle link would be slightly over center, held in place by the non-backdriveable worm gear. In addition, a detent and spring-loaded ball plunger is provided on the worm gear shaft to prevent the possibility of launch vibration loosening the worm gear.

To remove the baseplate from the receptacle the worm gear is reversed. The hooks are lowered and the cam on the linkage engages the roller on the receptacle pins. This provides positive disengagement of the connectors and seption of the bseplate and recepta

The latch mechanisms are installed in aluminum housings on each side of the module. The housings are connected by two torque tubes; an inner torque tube to actuate the latch mechanism, and a 4-inch diameter 0.090 in. wall outer tube to carry structural loads. The length of these tubes can be varied to accommodate modules of various sizes. Bolt holes and helicoils are provided to attach the module to the mechanism housings. The baseplate and receptacles are designed to accommodate modules up to 600 pounds and withstand the 10g crash loads. Side-load moments are reacted by outrigger pads on the worm gear end of the baseplate.

The bottom-mounted SRU interface mechanism engineering test unit shown in Figure V-11 was fabricated. The weight of the baseplate is 22 pounds and the weight of the baseplate receptacle is 5 pounds. This could be reduced for flight hardware. The mechanism was designed to be operated by the end effector used with the side-mounted interface mechanism. This end effector slightly

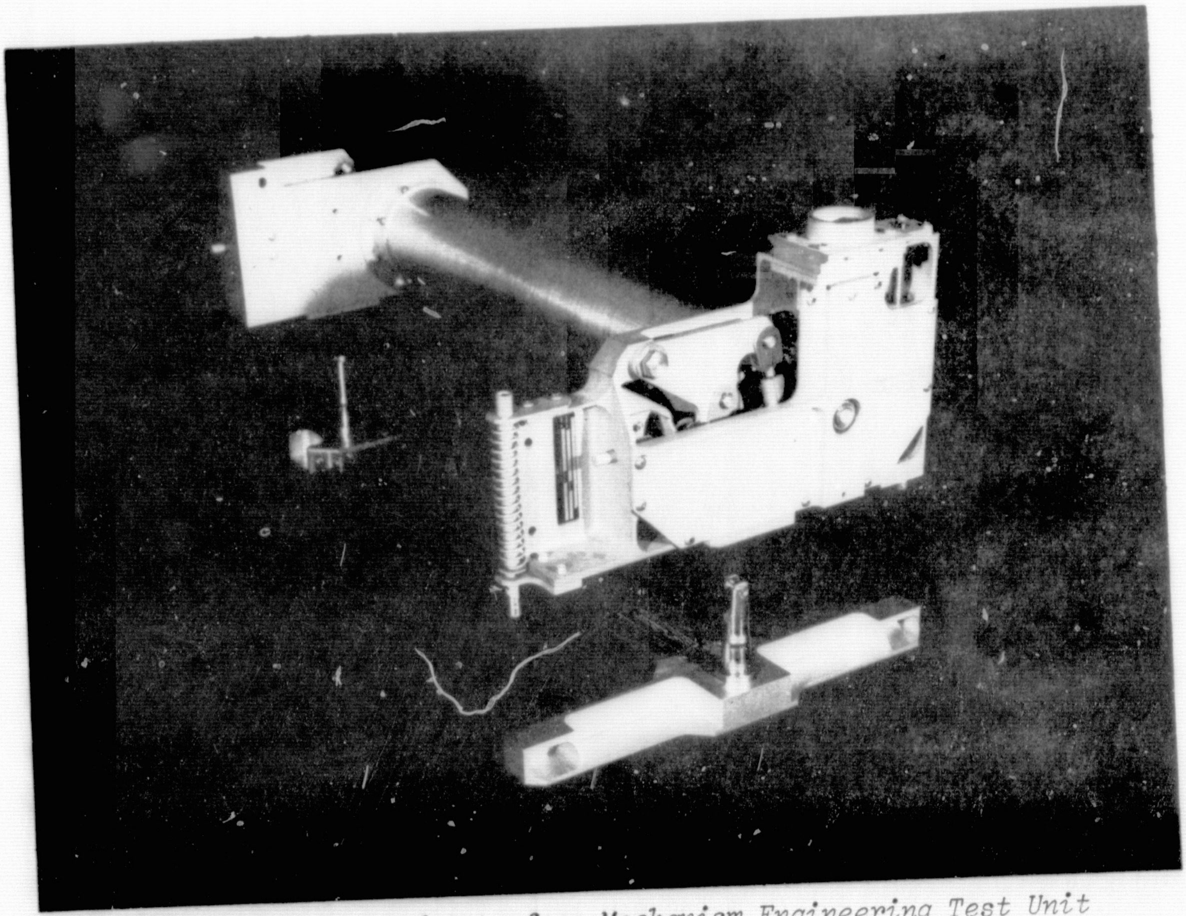


Figure V-11 Bottom-Mounted Interface Mechanism Engineering Test Unit

violates the module envelope. For flight hardware the end effector would be redesigned to eliminate this interference by rotating the plane of operation of the jaws by 90 degrees. This orientation would also be compatible with a minor redesign of the side-mounting interface mechanism.

5. Side-Mounted Interface Mechanism

A design of a side-mounted SRU interface mechanism was prepared. It was based on the Grumman EOS and TRW DSCS II concepts, meets all the system and element level guidelines, and uses only the good and acceptable mechanisms. It is adaptable over a range of SRU sizes from a 15-in. cube to a 40-in. cube. An engineering test unit has been fabricated which is suitable for modules in the 26 inch range.

The design is shown in Fig. V-12. The module mounts to the baseplate and the baseplate receptacle mounts to the spacecraft. Connector locations for

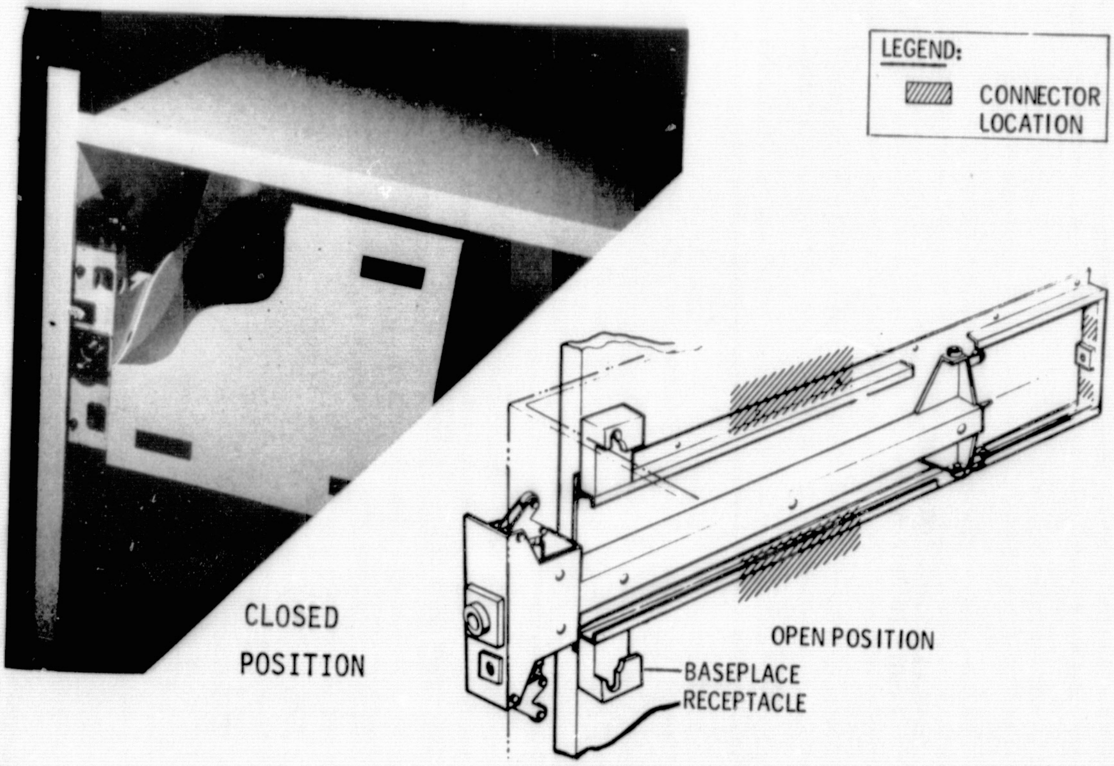


Figure V-12 Side-Mounting Interface Mechanism

the baseplate receptacle are shown. Corresponding locations are to be used on the baseplate. The baseplate mechanism details are shown in Fig. V-13. This mechanism is a rotating latch linkage device. On insertion of the module, the latch link rotates into the shaped, fixed latch catch assembly, firmly fastening the module to the spacecraft. When the link is reversed the module is forcibly separated from the spacecraft. The link roller eliminates sliding friction. The link is actuated by push rods which are moved by the bell crank that is turned by the worm gear. The worm gear is a 60:1 ratio set. If 27.6 in-lbs of torque (end effector output) is applied on the worm, initial latch link forces will be between 200 and 300 lbs. This force is more than adequate to force electrical connector halves together. As the bell crank approaches center position, these forces increase many times which ensures positive final close. The bell crank will be allowed to pass slightly over center for positive lockup. In addition, the non-backdriveable worm gear aids in maintaining latch-up. In the event that during launch, vibrations would cause this mechanism to back off and loosen, a safety latch is provided on the worm shaft to ensure against module loosening. This safety device is a spring loaded ball, laying in the detented worm shaft ring. The 60:1 gear ratio allows for the detents to be widely spaced.

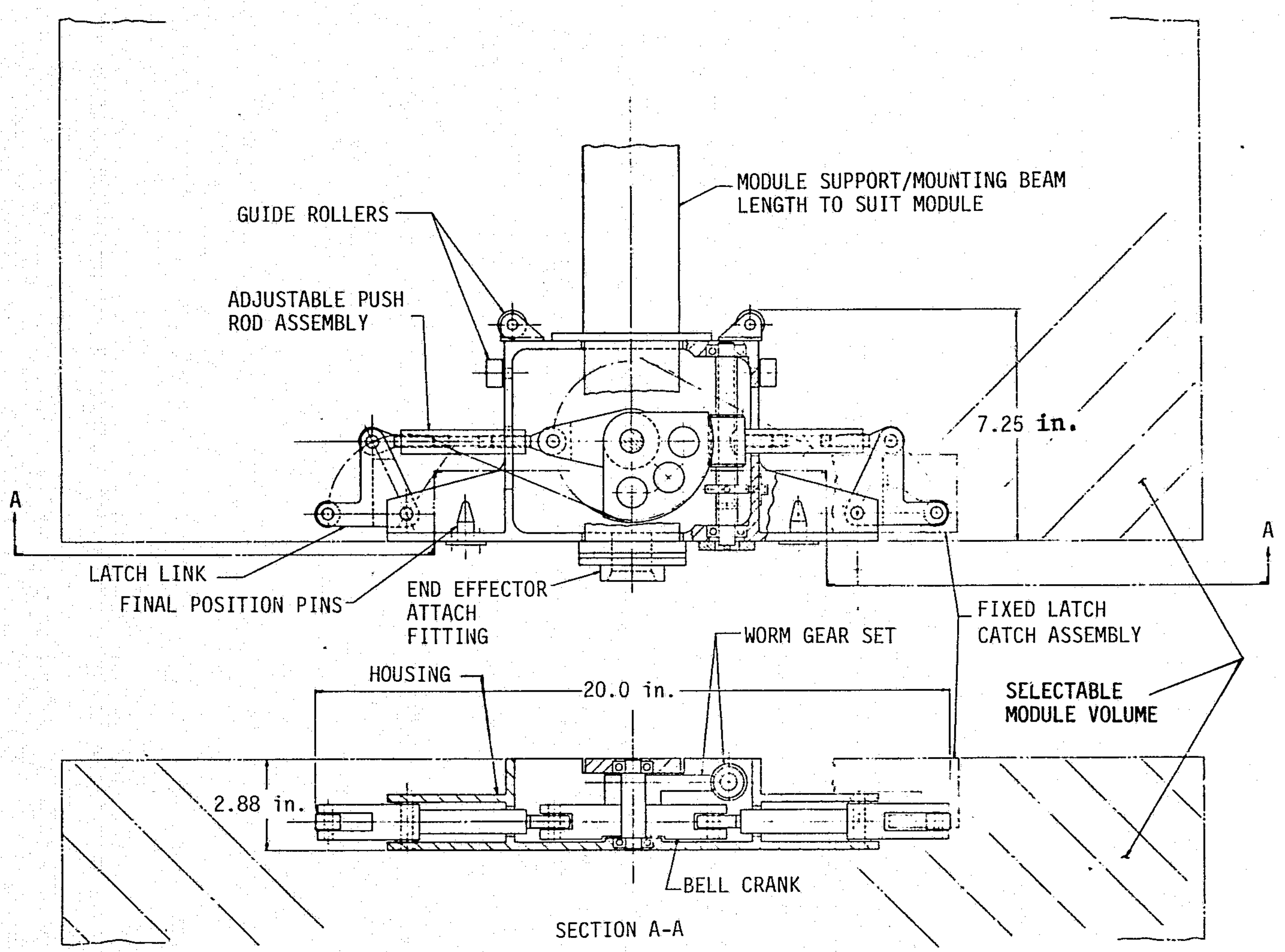


Figure V-13 Baseplate Mechanism Details - Side Mounting

Final positioning pins are fully engaged in the mating structure in the last 0.3 in. of module travel. The latch links are designed to provide a total final module movement of 1.75 in. This is considered to be adequate final closing movement for electrical connector engagement, and final alignment.

Guide rollers are mounted on the housing to guide the module in the rails and to provide initial alignment prior to final latch-up and alignment.

This mechanism is supported in a one-piece aluminum housing and would be supplied to the spacecraft module designer with a support mounting beam of a length suitable for that particular module. Mounting holes will be provided in the mechanism housing and support beam which the spacecraft module designer can use for mounting of module equipment as he desires.

The baseplate mechanism is supplied with a selectable length support beam. This 3 in. square tube with an 0.125 in. wall is intended to act as a beam and torsional member as module loading dictates. From this base structure (support beam and mechanism housing), the spacecraft module designer can add a floor, shear panels, and equipment support legs as required. The module size may be varied to occupy any volume depending on the module mass. The module may be flat or cubic in shape.

This mechanism is designed to support the heaviest module identified in the Aerospace Corporation operations analysis study, the 600-lb accelerometer module, under crash conditions of 10 g's. For lighter modules, it would seem that a lighter mechanism would be in order. This weight may be reduced 50% for the lightweight modules. It is not practical to design numerous sizes of latching mechanisms regardless of mechanism design, but it may be feasible to provide possibly two sizes, while maintaining the interface dimensions. If just this one size is used, it does not appear to be a great weight penalty as the mechanism is not overly heavy as designed.

The baseplate as shown here is estimated to weigh 14 lbs, which can be reduced for a flight unit.

The baseplate receptacle is shown on Fig. V-12 and consists of a prefabricated assembly containing the guide rails, final alignment pin receptacles, and the fixed latch catch fittings. These items are all prefabricated and supplied to the spacecraft designer as a unit which mounts directly into the

spacecraft structure. The spacecraft designer is not required to locate any matching elements of the latching mechanism. This is the kit concept. Guide rail lengths can be supplied to suit any module size.

The final module locating pins are positioned and are adequately sized to allow a considerable latitude in the spacecraft module shapes, sizes, and masses accommodated.

The engineering test unit fabricated to this design is shown in Fig. V-14.

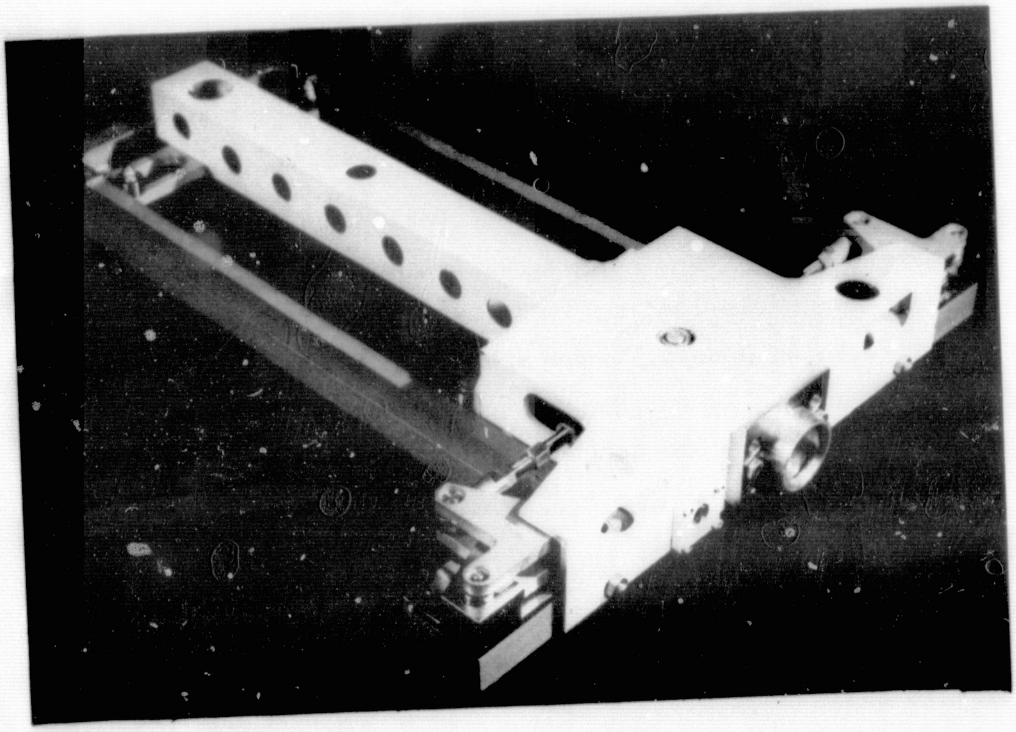


Figure V-14 Side-Mounting Interface Mechanism Engineering Test Unit

It reflects some changes necessary to meet the tight delivery schedule but reflects the important factors of the above preliminary design.

6. On-Orbit Servicer End Effector

The end effector concept is an extension of our prior work on general purpose manipulators and is shown in Fig. V-15. This end effector is designed to mate with either of the two interface mechanisms discussed above. It accomplishes two things: (1) it attaches the servicer mechanism whether it be a pivoting arm, general purpose manipulator, SRMS, or astronaut activated tool to the module; (2) it operates the latching mechanism. End effector attachment is accomplished by two closing jaws grasping the rectangular-shaped

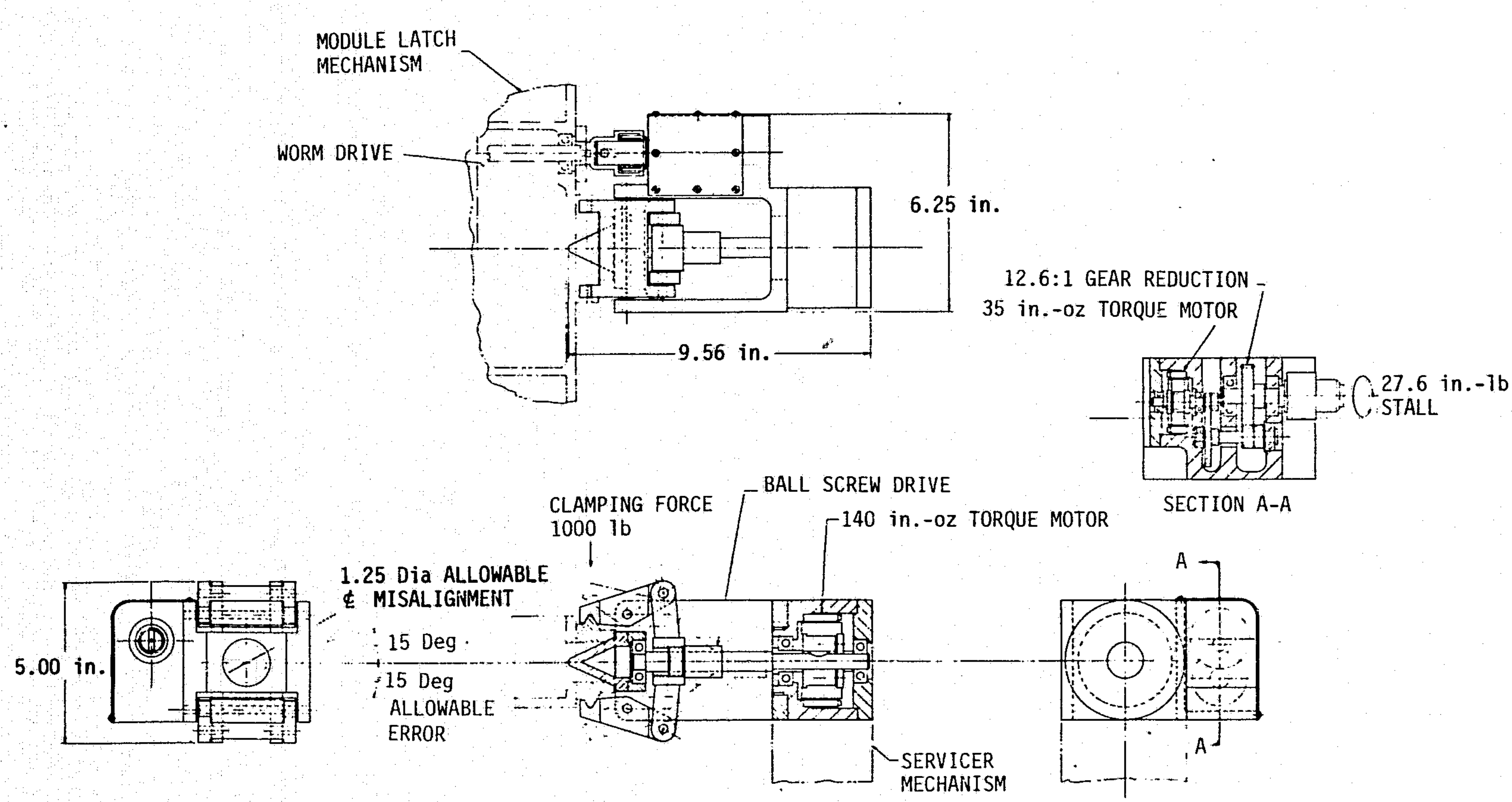


Figure V-15 On-Orbit Servicer End Effector

baseplate grip. The closing force is supplied by a motor-driven ball screw drive. This drive applies a low initial closing force when radial alignment is taking place and a very high final closing force when module handling is taking place. This high force occurs because the jaw links are approaching an over-center position with respect to the ball screw carriage. This operation will require no more than 5 seconds to close and should open much faster depending on the selected motor input current.

As the end effector is being positioned into the baseplate, misalignments are inevitable. The magnitude of these misalignments depends on docking position accuracies, control system accuracies, and manufacturing tolerances. These errors are not yet clearly defined, therefore it was decided to incorporate some reasonable allowable misalignment requirements at this time. As shown in the drawing the end effector may approach the module misaligned with a 1.25 in. diameter circle plus an angular error of ±15 deg. As the end effector approaches, the alignment cone engages the baseplate, forcing the end effector to align axially with the module. When the end effector jaws close, the end effector will be forced to rotate, eliminating the angular misalignment. As the end effector rotates, the baseplate latch drive centerline is automatically aligned with the baseplate latch mechanism drive shaft. The spring-loaded drive head falls into position over the drive screw. Engagement occurs when the drive is actuated. The drive mechanism is an integral part of the end effector attach drive. It is operated by a set of spur gears driven by an electric motor. The motor and gear train are designed to produce an operating torque of 27.6 in.-lbs with a stall torque of about 50 in.-lbs. Total average module latch-up (or unlatch) time should not exceed 15 seconds.

This device is fabricated as a one-piece aluminum housing with motor housing attached. No close tolerance gear requirements are necessary. The total weight is approximately 14 lbs. The attach part of this device may be used as a general purpose manipulator end effector.

The engineering test unit fabricated to this design is shown in Fig. V-16. It was necessary to substitute off-the-shelf motors to meet the delivery schedule and to make other appropriate changes. It was designed to operate with the side-mounted interface mechanism and it adapted readily to operation with the bottom-mounted interface mechanism.

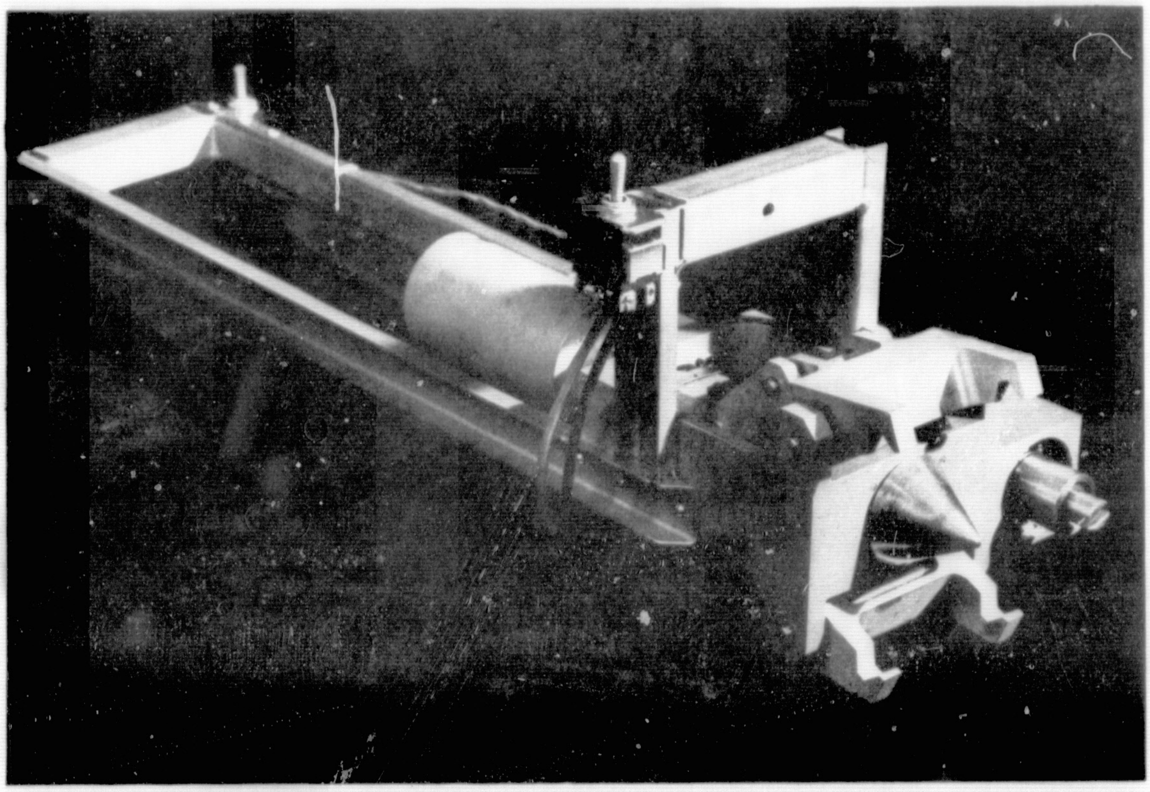


Figure V-16 End Effector Engineering Test Unit

The handle was provided to aid in demonstrations of the three engineering test units. If the end effector was to be used by an astronaut on EVA then a somewhat different style of handle might evolve. The switch nearest the jaws operates the jaws while the second switch at the left hand side of the figure controls the latch/unlatch motor.

D. RENDEZVOUS AND DOCKING

Rendezvous and docking is not part of module exchange on-orbit per se, but it is a necessary prerequisite for the performance of orbital maintenance and does have an influence on the design of the spacecraft and the on-orbit servicer, and can impact the STS. Its successful accomplishment is necessary for on-orbit servicing, but if the docking is unsuccessful, the spacecraft might be ruined. So the operating techniques at docking and the failure modes are important to the using spacecraft program.

Rendezvous and docking do influence spacecraft and servicer designs and there are a number of aspects that must be worked, but each of them only require normal engineering development. Assurance to the user that rendezvous and docking will be operational and safe must be developed as soon as possible to help obtain early user acceptance of on-orbit maintenance.

Rendezvous and docking are involved when the carrier vehicle and spacecraft come together for retrieval or for servicing. In many cases the carrier vehicle must also rendezvous and dock with the orbiter or tug. Candidate carrier vehicles are (1) full capability tug, (2) free-flying geosynchronous servicer, (3) solar electric propulsion system, (4) earth orbital teleoperator system, (5) some forms of the interim upper stage, and (6) the orbiter. These specific cases may be represented by two situations. First is the tug, with servicer, performing a rendezvous and docking with a spacecraft in geosynchronous orbit. This is the example case used here. The second situation is the tug returning to the orbiter where the SRMS is used for docking and berthing the tug. All of the expected combinations of spacecraft, carrier vehicle, on-orbit servicer, and orbiter can be represented by these two situations. Rendezvous involves the same general elements in both situations, while docking with the SRMS is different from using the carrier vehicle as the active element in docking.

The techniques of rendezvous and docking have been under study for many years. The United States and the USSR have each performed many

manned rendezvous and docking operations including a recent case involving spacecraft from both countries. The USSR has also performed rendezvous and docking remotely. The rendezvous and dockings for on-orbit servicing will generally be performed unmanned. The techniques of rendezvous are such that even the manned cases are conducted almost as if the spacecraft were unmanned, so the step to unmanned rendezvous is not major. Docking involves determination of relative position and attitude which has been done manually using visual aids. Many techniques have been proposed, built, and ground tested to perform docking automatically. These tests give confidence that automatic docking is possible, but the US has never, to our knowledge, performed an unmanned docking in space. However, unmanned docking is to be an STS capability and is being actively worked with every expectation that it will be an operational technique when required for on-orbit maintenance. The dockings at the orbiter are performed in a manned mode; it is only those at a distance from the orbiter that will be unmanned.

1. Representative Profiles

Representative profiles for rendezvous and for docking are presented. The rendezvous profile discussed involves flying a tug from the orbiter low earth orbit to mate with a spacecraft previously placed in geosynchronous equatorial orbit. The two most apparent divisions of the tug rendezvous mission are (1) ascent from the orbiter to the spacecraft and (2) descent from the spacecraft to the orbiter. The ascent part of the profile is discussed. The descent requires no flight mechanics techniques or hardware not discussed in the ascent phase.

The ascent phase itself is handled by separating it into two phases; an ascent to a gross targeting point and the terminal phase, from the gross targeting point to actual docking with the spacecraft.

The gross target point of ascent rendezvous is not taken to be the actual rendezvous point. There are two reasons: first, it is virtually impossible to simultaneously control the state variables of position and velocity vectors (at predicted time) without having throttable main engines or a bang-bang equivalent; second, the inaccuracies of the tug's

knowledge of its own position and velocity and of the target spacecraft's expected location do not permit prediction of the tug's given location relative to the spacecraft thousands of miles ahead at termination of final main engine burn. For these reasons, a target point is usually chosen to be biased off in such a way that regardless of dispersions and inaccuracies, the two vehicles, tug and spacecraft, will drift toward each other.

The terminal phase is begun after the tug reaches the gross target point. From this time on the tug guides itself with respect to the actual target vehicle which it literally senses; whereas in the prior phase of flight, the tug guided itself to achieve a computer-generated position and velocity target point.

The ascent rendezvous profile is shown in Fig. V-17 where the major points are noted on the figure. The nominal ascent from points 2 to 3 takes about 5.3 hours when no phasing orbit is used. Normally, phasing orbits will be used to establish the proper longitude relationship between the tug and spacecraft. The apogee altitude and number of phasing orbits are selected to provide the required phasing time which can be up to 12 hours. Each of the maneuvers of the figure is a tug main engine burn and is also used to accomplish part of the plane change between the orbiter orbit (usually 28.5 deg) and the spacecraft orbit (essentially zero deg). The majority of the plane change occurs at point 3. Rendezvous will be complete essentially one quarter orbit after point 3 (6 hours). The approach to targeting and determining the guidance presetting for the geosynchronous mission is baselined to be computed onboard the tug and completely independent of the tug deployment time, deployment orbit, or orbital inclination (within $\pm$ 1.0 degrees) which will decouple the orbiter operations from the tug operations. A midcourse correction burn is also used to help reduce dispersions at the targeting point.

A docking profile is shown in the upper half of Fig. V-18. The tug must locate the docking port on the spacecraft and then fly to an aimpoint which lies near the normal from the spacecraft docking port. The docking port can be located from transponders located on the spacecraft exterior or from prior knowledge of the spacecraft attitude orientation. Once the

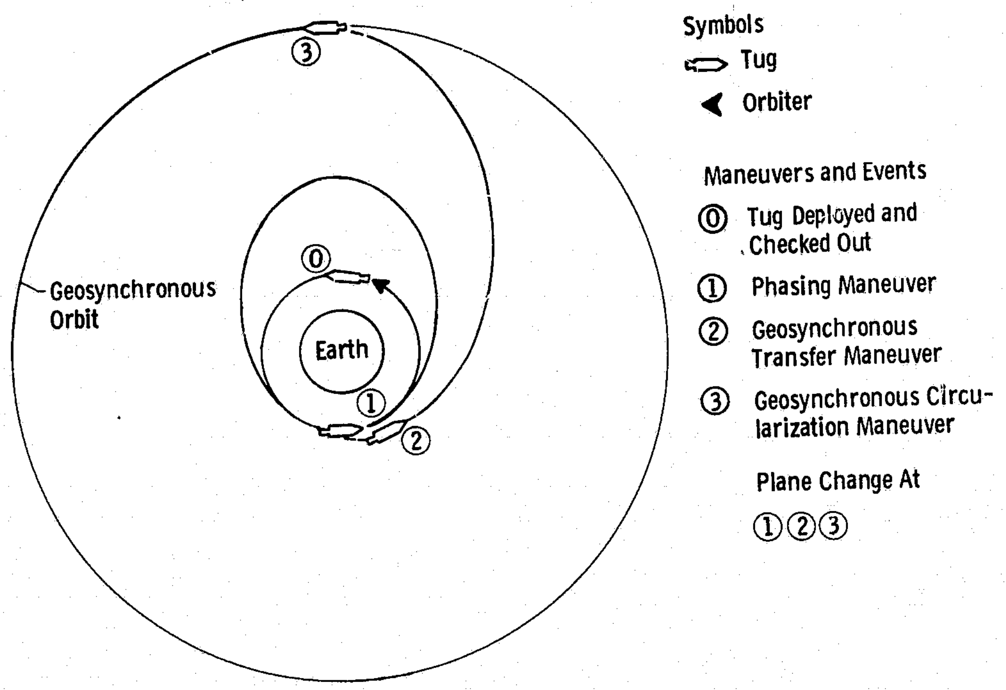


Figure V-17 Space Tug Rendezvous Profile

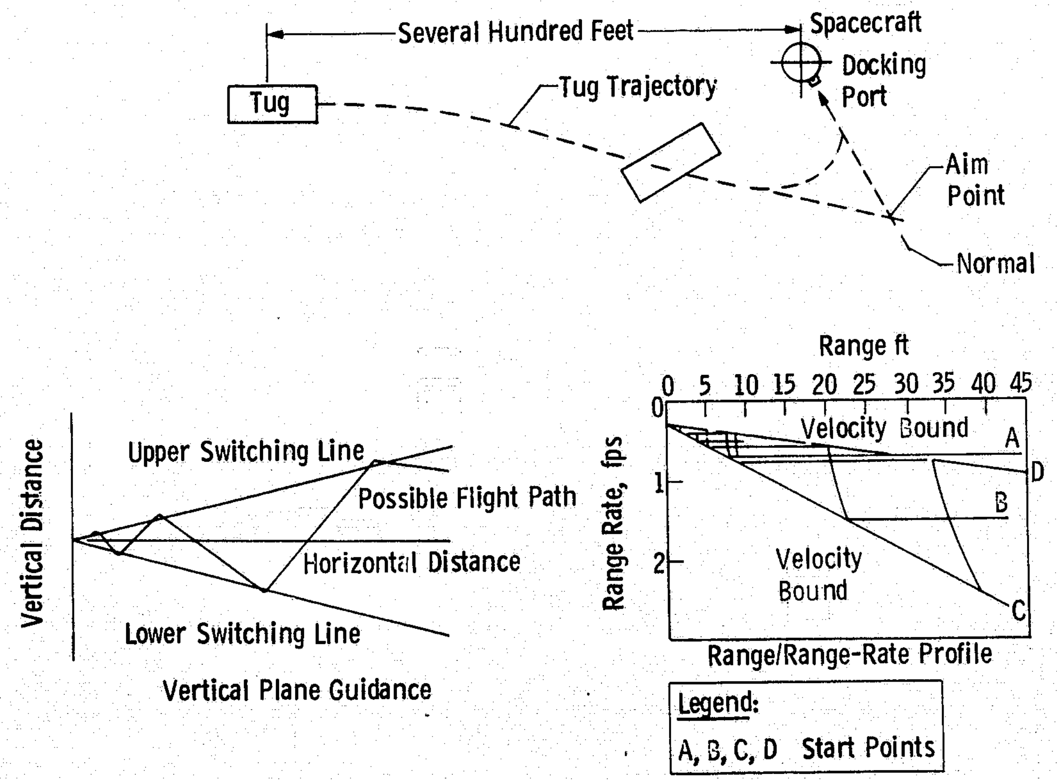


Figure V-18 Docking Profile

tug is near the aim point it will use an array of transponders located near the docking port. These short range transponders can be active or passive and are discussed below. The lower parts of the figure show typical guidance schemes that might be used from the aim point to docking. These were used during a Martin Marietta docking simulation in 1964 using active X-band radar hardware carried by a six DOF moving base simulator. The vertical plane guidance used two angles as bounds so that motion was loosely constrained at longer distances and became tightly constrained at the contact point. Similar angle bounds were used in the lateral plane. The lower right hand figure shows how the range rate is upper and lower bounded to higher velocities at longer ranges and lower velocities at the point of contact. Note that contact velocities can be made very low. These guidance laws are very simple yet excellent performance was obtained (less than one inch radial error at contact). The important part is the selection of sensors/transponders and contingency logic. The docking operation requires that the tug have full six axis control--translation and rotation--with minimum coupling between the two modes.

Once the tug and spacecraft are loosely docked together, then the docking mechanism is operated to draw the two together and to fasten them firmly. At this point the attitude control system of the spacecraft must be shut off. Also, provisions must be established to shut off spacecraft electrical power to those modules which are to be replaced. These functions can be provided via an electrical connector or a radio communications link.

While normally not necessary, it should be possible to keep a spacecraft, such as a communications mission, operating during on-orbit servicing involving the replacement of redundant modules. The docking impact can be kept small and if its direction is aligned well with respect to the spacecraft center of mass, then the spacecraft attitude can be kept within bounds. The tug could then maintain the desired spacecraft attitude during module exchange.

2. Rendezvous and Docking Equipment Required

The equipment required by the tug and spacecraft for rendezvous and docking are listed in Table V-8. The tug is considered to be the active vehicle with the spacecraft only required to hold relatively constant attitude

during rendezvous and docking. The additional items for docking are associated with the docking itself and with the relative attitude information. The guidance laws generally use position and rate of change of position information so the rate information must be computed or additional sensors provided.

A variety of manufacturers provide equipment which can satisfy the needs of Table V-8. They are too many to list here and the selection of the proper set is a part of the tug development process.

Table V-8 Rendezvous and Docking Equipment Required

VEHICLE	RENDEZVOUS	DOCKING
TUG	LONG RANGE TRACKING DEVICE	SHORT RANGE TRACKING DEVICE
	ATTITUDE CONTROL	ATTITUDE SENSORS
	ATTITUDE SENSORS	RELATIVE ATTITUDE SENSOR
	LINE OF SIGHT RATE SENSORS	SIX DOF REACTION CONTROL SYSTEM
	MAIN ENGINE	COMPUTER
	COMPUTER	DOCKING MECHANISM - ACTIVE
		LATCHING SYSTEM - ACTIVE
SPACECRAFT	TRANSPONDER	TRANSPONDERS
	ATTITUDE CONTROL	ATTITUDE CONTROL
	ATTITUDE SENSORS	ATTITUDE SENSORS
		DOCKING MECHANISM - PASSIVE
		LATCHING MECHANISM - PASSIVE

3. System Options at the Spacecraft

The rendezvous and docking system equipment explicitly required on the spacecraft, with alternatives in each class, are given in Table V-9. It is desirable that the spacecraft equipment be passive to decrease its probability of failure and thus reduce spacecraft losses due to this factor. However, use of active transponders increases the probability of acquisition at longer ranges while permitting reduced power in the tug tracking systems. Acquisition times tend to be long with the narrow beam laser systems.

Table V-9 Rendezvous and Docking System Options at Spacecraft

DOCKING MECHANISMS
Central or Peripheral
Energy Absorbing or Enveloping
LONG RANGE TRANSPONDERS
None - Skin Tracking
Corner Cubes
Flashing Lights
Active Transponders
SHORT RANGE TRANSPONDERS
Paint Patterns (TV)
Array of Corner Cubes
Array of Mirrors
Array of Flashing Lights
Visual Aids (TV)
Active Tracker and Data Relay

While there are many approaches to obtaining relative position and attitude data during docking with passive systems on the spacecraft, no single one has yet been selected. The selection will be part of the tug development process. The spacecraft transponder equipment tends to be light and small, both passive and active, and should not be a burden on the spacecraft. The passive docking systems are relatively light, (on the order of 30 lbs) and moderate in size (20 inch diameter for the central form).

4. <u>Impacts on Servicing Activity</u>

The equipment for rendezvous and docking tends to be located on the same spacecraft face as are the replaceable modules. Similarly for the on-orbit servicer stowage rack. The rendezvous and docking equipment could interfere with the servicing operations if it is not carefully located. Possible impacts are listed in Table V-10.

As is shown in Chapter VIII, the docking mechanism can intrude into the module exchange volume and thus reduce the usable spacecraft volume for module location. The docking alignment accuracy can complicate the control system as is discussed in Chapter VI. The sensors and transponders and their fields of view can be selected small enough and can be located so they need not interfere with module exchange.

Table V-10 Rendezvous and Docking System Impacts on Servicing Activity

DOCKING SYSTEM INTERFACE WITH MODULE EXCHANGE
DOCKING ALIGNMENT ACCURACY
ENERGY ABSORPTION SYSTEM STROKE
PROBABILITY OF SPACECRAFT DAMAGE
CONTAMINATION
STIFFNESS OF DOCKED AND LATCHED CONFIGURATION
TRANSPONDER SIZES, SHAPES, FIELD-OF-VIEW, AND LOCATIONS
SENSOR SIZES, SHAPES, FIELD-OF-VIEW, AND LOCATIONS
ELECTRICAL CONNECTIONS

In a system which involves a mechanical connection between two objects that are initially moving relative to each other, there is a possibility of unintended contact with resultant damage. The Apollo docking experience showed that damage could be avoided. However, one must be aware of the possibility and plan carefully on how to avoid it. The user community may well require analyses, tests, and space demonstrations to assure themselves that the probability of unintentional damage is very low. However, each of these potential impacts can be overcome by careful design of the rendezvous and docking system, the spacecraft, and the on-orbit servicer.

Control of the rendezvous operation can be automatic with verification to the ground. The midcourse correction can be calculated on the ground or on-orbit. Control of the docking phase is more complex because of the ramifications of a failure. The alternatives are similar to those discussed in Chapter VI with respect to the control of module exchange. A combination of supervisory control for the primary mode and remotely manned control for contingencies may be appropriate.

5. Rendezvous and Docking Impacts on the STS

No new rendezvous and docking impacts have been identified. The range of combinations of vehicles involved has been expanded but the techniques to be developed should be applicable for all of the new combinations. The

selected technique must minimize the probability of spacecraft damage. There must be an ability to reattempt after a missed docking has disturbed the spacecraft attitude. Provisions must be made for undocking even if the primary undocking system should fail. The advantages of multiple spacecraft servicing are such that the capability for repeated dockings on each mission should be developed. Each of these items can be considered to be part of the normal development of a rendezvous and docking capability for the tug and thus are not significant impacts on the STS.

VI. ON-ORBIT SERVICER CONTROL ISSUES

The control system selected to be used with the on-orbit servicer mechanism can strongly affect the servicer's operational utility and its versatility. If the control system is too limited in its capability, then so will the on-orbit servicer system be limited. Conversely, the control system need not have a comparatively greater capability than the servicer mechanism. Because of this importance, and because there was little in the literature on on-orbit servicer control systems, a top-level analysis of the situation was conducted with the objective of recommending a control system approach.

The task was performed in four phases: (1) review servicing concepts to identify control system requirements, (2) identify and analyze control system alternatives, (3) select the more useful control alternatives, and (4) recommend an approach for on-orbit servicer control development. The selected approach is to use supervisory control as the primary mode and remotely manned control to provide backup operation for failures and operational contingencies.

The selected system combines the best qualities of each of the two modes, and thus overcomes each of their deficiencies by using supervisory control as the primary mode and remotely manned control to provide backup operation for failures and operational contingencies. Because the remotely manned control is only a backup mode and will not be used frequently, longer operating times can be accepted. This permits use of a simplified TV camera(s) with very low frame rates (say three per minute) as well as using the TV system instead of proximity sensors for the alternative hazard avoidance system in this backup mode. Tolerance compensation can be handled by the operator using his ground-based computer. The major advantage of this combined mode is the availability of different and completely separate backup functions to obtain the highest probability of successful module exchange over the widest range of operating conditions.

A. CONTROL SYSTEM REQUIREMENTS

The general task from which the control system requirements were derived is (1) unstow the servicer mechanism, (2) remove a failed module from the spacecraft, (3) stow the failed module (temporarily or permanently), (4) locate the replacement module in the storage rack, (5) place

the replacement module in the spacecraft, (6) verify that the replaced module is latched in place, (7) permanently stow the failed module, (8) repeat as necessary, and (9) restow the servicer mechanism. Note that rendezvous and docking and spacecraft checkout are not considered in this particular analysis.

The servicer control system requirements were developed from the above general task outline, the literature referenced, and consideration of the pivoting arm and the general-purpose manipulator forms of servicing mechanisms. These requirements are listed in Table VI-1 and should be applicable to most servicer mechanism configurations. The usual requirements of system stability, low weight, and low life-cycle cost would also apply.

Table VI-1 Control System Requirements

Exchange Modules One at a Time
Accommodate a Variety of Module Sizes, Masses, Locations, and Orientations
Operate with Different Spacecraft on a Single Mission
Up to 25 Module Exchanges per Flight
Provide Backup Modes
Generate Signals for Individual Mechanism Joints
Provide Required Accuracy and Repeatability
Compensate for All System Tolerances
Avoid Control Anomalies (e.g., Singular Points)
Provide for Hazard Avoidance
Provide Suitable Stiffness
Be Compatible with Structural Flexibility
Minimize System Data Rates
Accommodate Data Transfer Delays

In general, it takes six degrees-of-freedom (DOF) to define the relative position and attitude of one object (e.g., module) with respect to another (e.g., spacecraft), yet we are suggesting the use of a four-DOF system (pivoting arm) to perform module exchange. The answer lies in an examination of the uncertainties that might occur in the uncontrolled degrees of freedom. The pivoting arm system can position a module anywhere on the end of a spacecraft, in any orientation, and over the required

range of axial positions. The unavailable degrees of freedom might then be called the two angles pitch and yaw. They are defined by the geometric uncertainties of the hard docking and spacecraft latching system. These location geometry uncertainties, in pitch and yaw, occur at the spacecraft periphery and thus can be well controlled because of the long baseline (spacecraft diameter). The resulting effect is that the mechanism must bend slightly (with a possible corresponding readjustment of the controlled variables) to make up for the misalignments. The bending directions for the pivoting arm are directions in which the mechanism tends to be naturally softer. For a representative set of numbers, the arm natural frequency when moving a module would need to be on the order of 1 Hz to be soft enough to accommodate the anticipated docking uncertainties in pitch and yaw.

When a module is being moved from the stowage rack to the spacecraft, the module will be turned end-for-end so the same set of module latches can be used for the spacecraft and stowage rack. If the modules are rotated about an individual mechanism axis, the space between the stowage rack and spacecraft must be greater than two module lengths. Conversely, if the mechanism coordinates its motions so the module is rotated about its geometric center, the spacecraft-to-stowage rack separation can be reduced to a little over one module length. A more careful evaluation of this situation when the mechanism end effector geometry is considered results in a slightly more complex trajectory and the need to provide additional clearance for the end effector. Thus module trajectories will generally require coordinated motions of the mechanisms.

Table VI-1 requires that the selected control system avoid control anomalies and, in particular, singular points where the desired motions cannot be obtained by the controlled variables. One aspect of this was discussed above with respect to limited degrees of freedom. For the general purpose manipulator with six DOF a similar situation exists near a line that is an extension of the first joint axis. The problem is overcome by proper orientation of the singular line so it does not pass through the work area or by use of a seven-DOF manipulator.

The major geometrical error anticipated is associated with roll about the docking system axis. This is sometimes called a clocking error. The error results from the fact that many docking systems have

not had to correct this error and it is difficult, though not impossible, to reduce. Note that both mechanisms have a control axis parallel to the docking system roll axis. Thus, if an acceptable way can be found to measure the clocking error, it can be readily biased out of the system.

Any positioning errors that cannot be removed from the system must be accommodated by a "capture area" at the mechanical interface. This capture area might typically be a one-inch diameter circle. When attachment is made, the positioning errors show up as bending in the mechanism and control system error signals that can generate control torques and motor overheating. This latter aspect can be alleviated by control loop gain reduction or the use of a rate control mode.

B. CONTROL SYSTEM ALTERNATIVES

Three general control modes will be addressed--automatic, supervisory control, and remotely manned. Each will be discussed in turn as to how it satisfies the requirements of on-orbit servicing. Servicing in geosynchronous orbit will be our example case though much of what is said also applies to operation in the orbiter cargo bay.

Our reviews of prior studies of orbital servicing uncovered little as to how module exchange could be controlled. Automatic control and supervisory control have been briefly mentioned for the geosynchronous case and remotely manned control for the orbiter cargo bay operations.

1. Automatic Control

The automatic control mode involves a control electronics assembly (CEA) that controls all the module exchange activities including module trajectories, hazard avoidance, sequencing, activity completion indications, redundancy, fail-safe aspects, and tolerance compensation. A block diagram is shown in Figure VI-1 where the spaceborne control equipment is shown above the dashed line and the ground- or orbiter-based equipment below. The CEA contains all the logic for control functions as shown. For automatic control, the various algorithms must be sophisticated enough to handle all anticipated failures and unanticipated contingencies. Fail operational/fail-safe operation implies at least double redundancy for all functions, with triple redundancy for functions required to allow the servicer to release from the spacecraft. The tolerance compensation algorithm uses measurements of docking errors and other

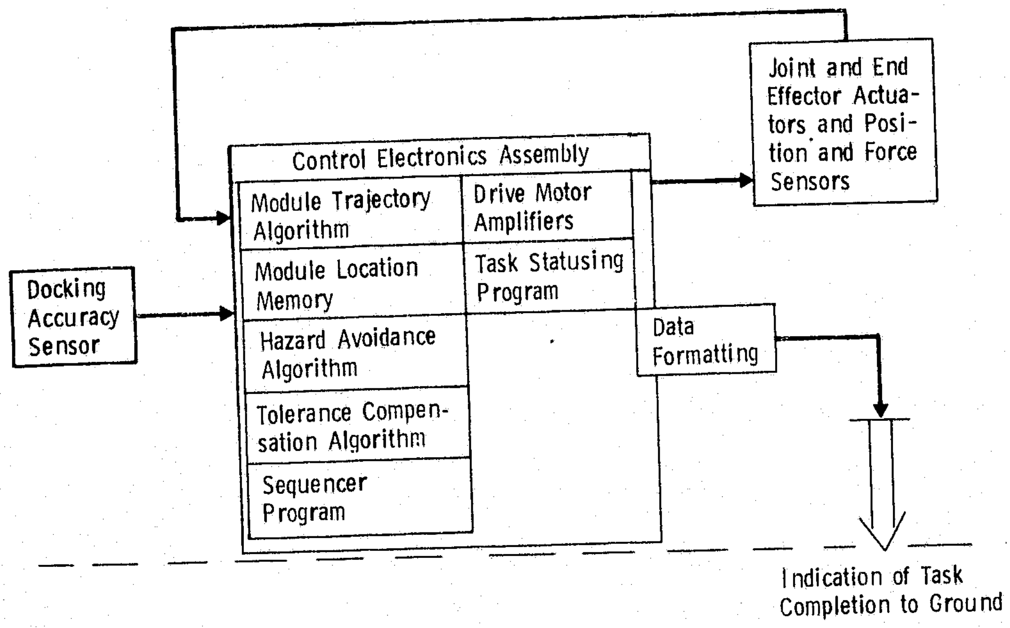


Figure VI-1 Automatic Control

significant system errors and biases the stored module locations appropriately. Module exchange would be along a trajectory that minimizes chances of the moving parts unintentionally contacting any structure. These trajectories could be developed on the ground using a simulator and then stored in the programmer. An alternative is to develop an algorithm that will solve for safe trajectories as a function of the end points. This algorithm would then be stored in the spaceborne CEA. A second alternative is to develop a set of standard "safe" subtrajectories; e.g., turn module end-for-end, and then have a set of connecting regions where any trajectory is safe. The total trajectory would consist of alternate standard and connecting trajectories. As all of the trajectories used in this way are safe, there would be no need to verify each selected trajectory. The tradeoff is between a one-time development and higher CEA costs versus solving the problem individually for each case. Hazard avoidance then is involved in safe trajectory generation, but not explicitly in the on-orbit operations.

The automatic mode can satisfy most of the Table VI-1 requirements. It particularly minimizes communication system data rates and the effects

of data transmission delays. The difficulties lie in accommodation of system tolerances and the approach to the fail-safe/backup mode considerations. The CEA and mechanism errors can generally be made small. The errors associated with docking, thermal effects, and manufacturing tolerances are larger. In particular, many docking systems are designed with poor control over vehicle relative roll angle (clocking errors) because these errors usually can be accepted. For module exchange, the errors can be important. These geometric uncertainties can be accommodated by (1) large mechanism end effector and module latch capture volumes, (2) reduced tolerances, (3) measuring the error and appropriately biasing the control system computations, or (4) providing a closed-loop tracking system that will cause the mechanism end effector and module latches to home in on the actual positions. Which of the four compensation techniques to use must depend on the results of a detailed tradeoff study. Note that the end effector-to-module and module latch-to-spacecraft (and stowage rack) capture problem is a close parallel to the tug-to-spacecraft docking problem. In the three cases all six relative degrees of freedom and their rates must be controlled based on some combination of measured and precalculated values.

Selection of the approach to meeting the fail-safe and backup requirements is much more difficult for the automatic mode. Fail-safe can be handled for many elements by triple redundancy. This approach is not as easy in the mechanical aspects. Design of an automatic system with the adaptability to handle unanticipated contingencies is quite difficult. The automatically operated decision logic that transfers from the primary mode to the backup mode also presents a design challenge. The need to prevent servicer malfunctions from inhibiting later repair of a spacecraft also presents a difficult design challenge for the automatic control mode. In brief, there appears to be no cost effective way to ensure fail-safe operation and to provide a backup mode for fully automatic operation.

The advantages of the automatic control system are in the low data rates to the ground and effective hazard avoidance in the primary mode. The disadvantages are in difficult contingency hazard avoidance, difficult tolerance compensation, and very limited alternatives for independent backup operation for failures and operational contingencies. An automatic control system with adaptive features for the required reliability will

be very expensive to implement.

2. Supervisory Control

The supervisory control mode involves a CEA that controls the detail module exchange activities, including module trajectories, hazard avoidance, and some of the redundancy aspects. Man, through a command and data link, selects sequencing, acknowledges completion, and provides some of the redundancy and fail-safe aspects. Supervisory control, as shown in Fig. VI-2, is a modification of automatic control with the module exchange initiation and acknowledgment controlled and displayed at the ground. The onboard CEA generates and controls the specific module trajectories after the operator has identified the locations of the modules to be replaced. To minimize communication system data rates, it is assumed that the supervisory mode does not involve a television system. Module exchange trajectories could be developed and used as previously discussed for the automatic mode. The only difference is that a subsequent servicer mechanism (module) trajectory could only be initiated by a discrete signal sent from the man. Because the operator would be allowed to initiate only those trajectories stored onboard, the hazard avoidance problem would be similar to that for the automatic mode.

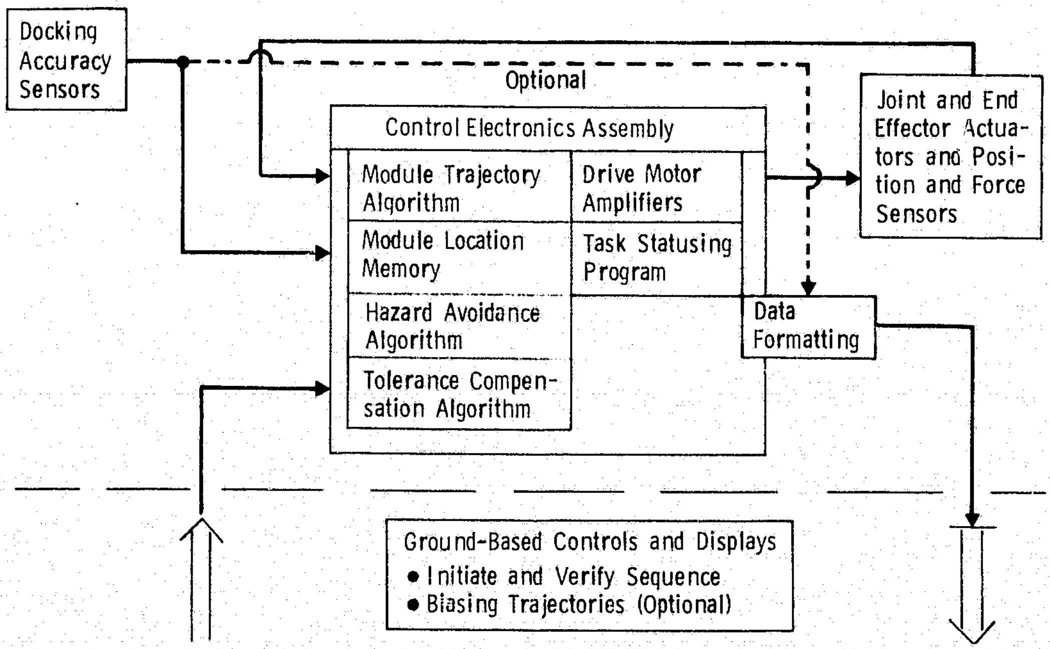


Figure VI-2 Supervisory Control

Because it is in many ways similar to the automatic mode, the supervisory control mode also can satisfy most of the Table VI-1 requirements. Supervisory control will require higher data rates and be slightly more affected by transmission time delays. Neither effect is very significant. However, supervisory control is similar to automatic control in accommodation of system tolerances. It is possible to involve man in the measurement of system errors and their introduction into the CEA biasing scheme. Man could also be involved in the closed-loop tracking by the mechanism end effector of the module attachment point. Either of these approaches requires that the proper data be collected and transmitted to the ground for use by the operator. The CEA is less complex for this mode than for the automatic mode because the module trajectory algorithm, module location memory, and tolerance compensation algorithm can all be modified from the ground to work around some failures and unanticipated contingencies.

With regard to the fail-safe and backup requirements, the supervisory control mode is no better than the automatic mode unless the operator is provided with more data and control path alternatives. If the operator is reduced to initiation of stored programs, his ability to act in a backup capacity is very limited.

Although this modification of automatic control introduces some ground based manual and computer backup control possibilities, there is a minor increase in communication system data rates and the basic limitations of automatic control are not adequately alleviated to provide sufficient backup operation for overcoming failures and operational contingencies.

3. Remotely Manned Control

The remotely manned mode involves an onboard CEA and sensor system, a two-way communication link, and an operator at a control and display station (C/DS). The operator controls all the module exchange activities including module trajectories, hazard avoidance, sequencing, activity completion acknowledgment, and fail-safe aspects. The redundancy aspects would still be a part of the machine.

The remotely manned control approach, as shown in Figure VI-3, takes nearly all the CEA functions to the ground and brings the cognitive and adaptive capabilities of man fully into play. The significant on-orbit additions are the TV camera(s) with high communication system data rate

requirements and the proximity sensors to provide a backup hazard avoidance function. An option to include the feedback of measured forces at the end effector in the form of an inner control loop (discussed below) shown here can become a part of any of the control modes.

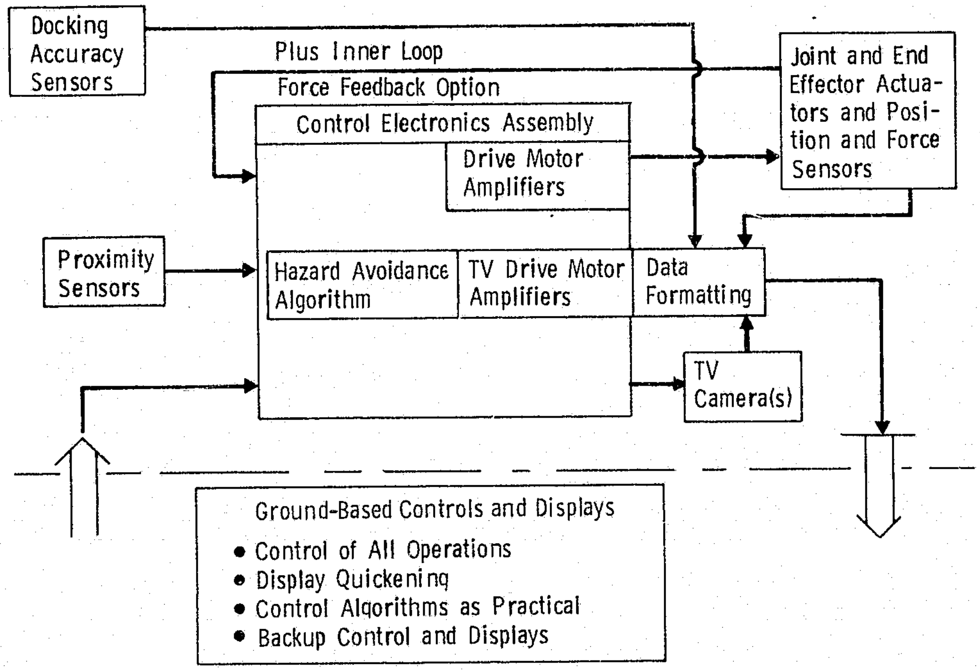


Figure VI-3 Remotely Manned Control

Trajectories would be generated in response to the operator's control inputs. Many control input devices have been suggested and used in the past such as (1) joint-by-joint, (2) replica, (3) nongeometric correspondence, (4) three-DOF hand controllers, and (5) six-DOF hand controllers. The relative value of these devices has been discussed extensively in the literature. The multiple-degree-of-freedom hand controllers with the same number of degrees of freedom as the exchanger mechanism appear suitable for this task. Use of a rate control mode seems suitable for the level of control and time available. Its use also avoids the need for reindexing systems. Coordination of display and hand controller motion is very desirable. Coordinate system transformations can be performed on the ground or on-orbit.

The possible need for outer-loop force-feedback control, where the operator "feels" the control forces being exerted, can also be discussed. However, simulations at Martin Marietta in late 1973 indicated that

effective manipulator arm control could be obtained in a rate mode without outer-loop force-feedback. For the levels of force in the simulated tasks, and in the tasks of module exchange, the forces being fed back are not significant to the operator. Since outer-loop force-feedback also requires a much wider communication system bandwidth, it is suggested that outer-loop force-feedback not be included unless later work shows it must be used. The use of inner-loop force-feedback, where the forces at the end effector are measured and signals are generated to reduce the forces (and moments, in some cases) not in the desired direction to zero, cannot be decided so easily. MIT has shown (Chapter XI, Item O-7) that long pins can be inserted into holes with very low clearances using this technique of minimizing lateral forces while maintaining an axial force. As all signal processing can be done onboard, the communication channel bandwidth is not affected. This force measurement might also be used in the hazard avoidance system. It is suggested that inner-loop force-feedback be considered as part of on-orbit servicing control for the remotely manned mode and possibly also for the automatic and supervisory control modes.

For most of our manipulator system investigations, we have used visual feedback, as have most other investigators. Television is the obvious sensor for orbital operations. The cameras are small, lightweight, relatively inexpensive, and very versatile. Their power requirement is also low, but their need for supplemental lighting must be evaluated for each situation. The TV system's main disadvantage is communication system bandwidth. This bandwidth requirement can be reduced by reducing frame rate, line pairs, and gray scale to those needed for module exchange. Stereo TV has been shown to be useful in some cases as have two cameras with orthogonal viewing. TV systems for servicing could also be used for spacecraft docking and for spacecraft inspection prior to docking. Camera and lighting location selection must be left to a careful evaluation of the operational environment.

Hazard avoidance in the remotely manned mode becomes difficult because the number of trajectories that might be commanded increases significantly. One alternative is to use the command signals, as derived from the operator's control input devices, combined with computer graphic techniques in a ground computer to generate bias signals that would prevent module collisions. The computer would inform the operator of near hazards and that

it was biasing the command signals. This open-loop form of hazard avoidance leaves much to be desired because its calculations may not be based on true data. Onboard alternatives would include use of proximity warning sensors or the inner-loop force-feedback sensors previously discussed. The TV cameras can also be used for hazard avoidance.

Although the remotely manned mode can satisfy many of the Table VI-1 requirements, it requires the highest communication system data rates and is most susceptible to data transfer delays. This susceptibility is particularly true for outer-loop force-feedback systems. The remotely manned system can best accommodate system tolerances because properly selected TV systems will accurately display the errors to be corrected. The system can be made mostly fail-safe by incorporating logic that will stop the mechanism whenever the control loop is broken or when the operator shuts off power. The operator can shut off mechanism power whenever he notes any unusual motion and thus provide another increment of fail-safe operation for the module motion part of the system activity. Should the servicer fail to release the spacecraft, the operator could initiate a pyrotechnic system that would ensure spacecraft separation.

Backup modes can be provided for remotely manned operation. Each function, except the mechanism itself, can be independently paralleled. For example, the servo feedback pots instead of the TV can provide position data. Two control/display stations and operators (or some equivalent engineering compromise) can also be provided, as can communication system redundancy.

The advantages of remotely manned control are the introduction of man's cognitive capability, a more complete backup capability, and good compensation for system tolerances. The penalties are very high data rates, an increased susceptibility to data time delays, and a more complex hazard avoidance situation because of the variety of trajectories the operator might generate in comparison with the preprogrammed trajectories of the automatic mode.

C. EVALUATION AND SELECTION

From the discussions of each of the three control systems, it is obvious that each has good areas and bad areas, with none appearing to completely satisfy all requirements. The three are now compared and analyzed

to identify a suitable compromise.

1. Evaluation

The above discussions of the three control modes concentrated on five of the requirements of Table VI-1. A more complete evaluation is required and some of the general factors not listed in Table VI-1 will be brought in. The following list of requirements can be met by proper design of any of the three control modes or their combinations: (1) exchange modules one at a time, (2) generate signals for individual mechanism joints, (3) provide required accuracy and repeatability, (4) avoid control anomalies, (5) provide suitable stiffness, and (6) be compatible with structural flexibility. Table VI-2 lists the factors used for further evaluation. The first five factors come from Table VI-1 and have been discussed. Note that hazard avoidance has been divided into the areas of prime and contingency or backup because the evaluation tends to be quite different for the two areas, yet both are important. The sixth item, capability, combines three items from Table VI-1: (1) accommodate a variety of module sizes, masses, locations, and orientations, (2) operate with different spacecraft on a single flight, and (3) make up to 25 module exchanges per flight. Versatility is used to bring in the need to be able to change and reorder things late in the launch preparation sequence or perhaps during the mission. Complexity of the spaceborne equipment and of the control station equipment are self-explanatory. When the control station is located in the orbiter, the penalty for a complex control station becomes relatively more significant. The next three items are also self-explanatory. The last item reflects the relative ease of providing the additional redundancy/backup at the control station that can be used to help provide control system redundancy/backup.

An examination of Table VI-2 confirms that none of the three modes will satisfy all requirements. The best system would be a combination of automatic and remotely manned but this is impossible by our definition of automatic. The equivalent thus becomes a combination of supervisory control and remotely manned control using the best parts of each.

Table VI-2 Control Mode Evaluation

EVALUATION PARAMETER	TYPE OF CONTROL: AUTOMATIC	SUPERVISORY	REMOTELY MANNED
Provide Backup Modes	Redundancy only	Redundancy + sequence modification	Different
Compensate for All System Tolerances	Difficult	Moderate	Easy
Implement Hazard Avoidance	Prime - inherent; Contingency - very difficult	Prime - inherent; Contingency - difficult	Prime - complex: Contingency - least difficult
Minimize System Data Rates	Minimal	Low	High
Accommodate Data Transfer Delays	Easiest	Easy	Most difficult
Capability	Lowest	Medium	Highest
Versatility	Lowest	Medium	Highest
Complexity - Spaceborne Equipment	Highest	High	Low
Complexity - Control Station	Least	Low	Highest
Operator Training	Very general	Mostly general	Highly specific
Risk of Damage Because of Operator Error	None	Very little	Very high
Reliability versus Cost	Very low	Low	Medium
Ground or Orbiter Redundancy/Backup	None	Some	More

2. Selected System

The selected system is a blend of supervisory and remotely manned control with the characteristics listed in Table VI-3. When the system operates in the prime mode, it is the supervisory control system we have described above except that the system errors are measured remotely by the ground operator and then biased in the onboard CEA. The ground operator provides the contingency and backup operation. A lower performance in terms of module exchange time is accepted in the backup mode which in turn

permits use of a low data rate TV system that has a picture refresh rate of perhaps three frames per minute.

Table VI-3 Recommended Control System Characteristics

Primary Mode - Supervisory Control
Backup Mode - Remotely Manned Control
Stored/Interpolated Module Trajectories
Hazard Avoidance
Supervisory - Precalculated
Remotely Manned - TV and Ground Computer Graphics
System Errors Measured by Man and Biased Onboard
Separate Translation and Rotation Hand Controllers
TV and Mechanism Position Displays
Mechanism Joint Control
Supervisory - Position
Remotely Manned - Rate
TV Refresh Rate - Three per minute

A block diagram of the recommended system is shown in Figure VI-4. It appears very similar to the supervisory control system of Figure VI-2, except for the contingency and backup modes, adding the TV cameras, and deleting the tolerance compensation algorithm. The TV system is used instead of proximity sensors for the alternative hazard avoidance system in the backup mode. One major advantage of this combined mode is the availability of different and completely separate backup functions to obtain the highest probability of successful module exchange. It also brings the cognitive and adaptive capabilities of man into the situation as they are needed without burdening him with the routine activities. Figure VI-4 does not show inner-loop force-feedback, but it certainly should be considered for this application. Another area for investigation is a careful examination of the TV requirements to determine if reductions in gray scale, resolution, or field of view can be accepted to further reduce communication system bandwidth.

A review of Table VI-2 for our selected system shows that a significant improvement in capability results for a moderate level of communication system data rates, system complexity, and operator training.

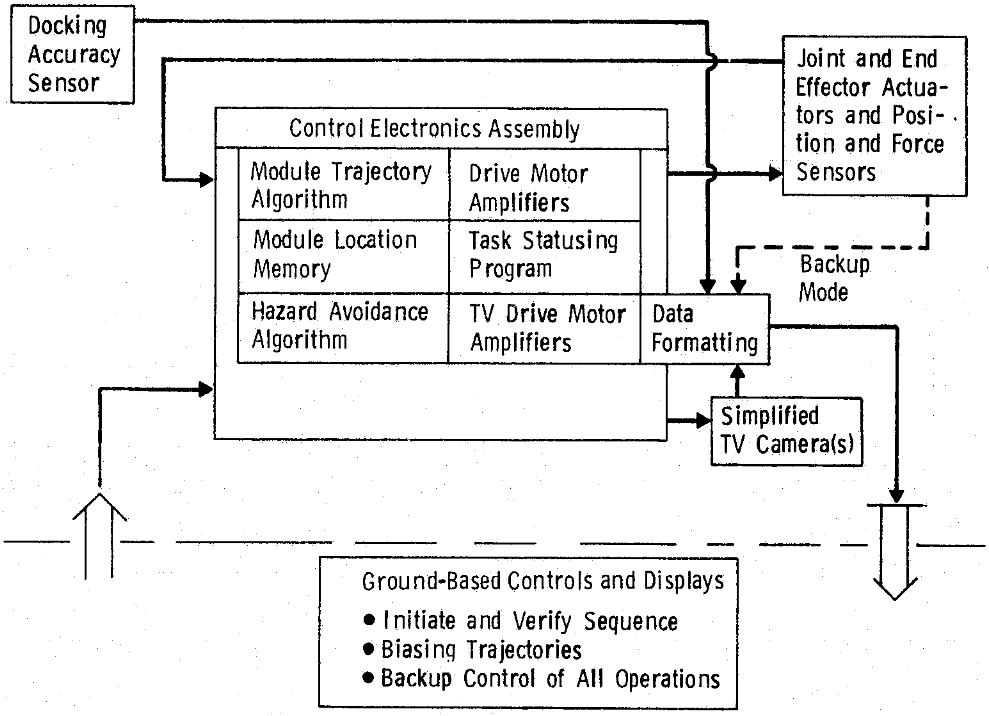


Figure VI-4 Supervisory with Remotely Manned Backup Control

In summary, the selected system combines the best qualities of each of the two modes, and thus overcomes each of their deficiencies by using supervisory control as the primary mode and remotely manned control to provide backup operation for failures and operational contingencies. Because the remotely manned control is only a backup mode and will not be used frequently, longer operating times can be accepted. This permits use of a simplified TV camera(s) with very low frame rates (say three per minute) as well as using the TV system instead of proximity sensors for the alternative hazard avoidance system in this backup mode. Tolerance compensation can be handled by the operator using his ground-based computer. The major advantage of this combined mode is the availability of different and completely separate backup functions to obtain the highest probability of successful module exchange over the widest range of operating conditions.

D. DEVELOPMENT PLAN

The importance of the on-orbit servicer's control system with respect to its utility and versatility when combined with the control system's low state of development implies the need for a comprehensive development plan.

The major items of a development plan are given in Table VI-4. The control modes must be adequately defined before they can be simulated, evaluated and compared. Simulation is a major tool in control system development. The specific items for evaluation in the simulations and analytical evaluations are given in Table VI-5.

Table VI-4 On-Orbiter Servicer Control System Development Items

DEFINE SELECTED CONTROL MODES TO A LEVEL SUFFICIENT FOR SIMULATION
CONDUCT SIMULATIONS AND ANALYTICAL EVALUATIONS
DETERMINE VALUE OF INNER-LOOP FORCE-FEEDBACK
DETERMINE MINIMUM ACCEPTABLE TV REQUIREMENTS
DEFINE GROUND SUPPORT REQUIREMENTS FOR MISSION OPERATIONS AND CONTROL
PERFORM DETAIL DESIGN OF SERVICER CONTROL SYSTEM FOR PROTOTYPE HARDWARE
IDENTIFY FLIGHT DEMONSTRATION REQUIREMENTS

Table VI-5 Simulation and Analytic Evaluation Considerations

LOGICAL DIVISION OF FUNCTIONS BETWEEN MAN AND MACHINE FOR PRIMARY AND BACKUP MODES
TRADEOFF BETWEEN ON-ORBIT AUTOMATION AND TELEMETRY DATA RATES
APPLICABILITY OF SUPERVISORY CONTROL MODES AND FORCE STEERING CONCEPTS
OPTIMUM COMMUNICATION SYSTEM BANDWIDTHS (e.g., TV FRAME RATE)
EFFECTS OF STRUCTURAL FLEXIBILITY AND TELEMETRY TIME DELAYS
METHODS TO COMPENSATE FOR STRUCTURAL, DOCKING, AND CONTROL SYSTEM TOLERANCES
HAZARD AVOIDANCE HARDWARE AND SOFTWARE REQUIREMENTS
MODULE TRAJECTORY SOFTWARE REQUIREMENTS
TASK STATUSING AND SEQUENCER SOFTWARE REQUIREMENTS
MODULE IDENTIFICATION AND LOCATION HARDWARE AND SOFTWARE REQUIREMENTS
APPLICATION OF COMPUTER GRAPHICS AND VIDEO GUIDANCE
CONTROL AND DISPLAY HARDWARE AND SOFTWARE REQUIREMENTS
POSITION AND FORCE FEEDBACK REQUIREMENTS
ADDITIONAL SRT REQUIREMENTS

The potential value of inner-loop force feedback, or force steering, was discussed above. It might well be particularly useful with the four DOF pivoting arm servicer mechanism that is recommended. The teleoperator visual system study performed by Martin Marietta for MSFC in 1973 (Chapter XI, Item J-21) indicated that control might be possible with drastically reduced bandwidths. This possibility should be investigated. The ground support requirements for mission operations and control should be investigated to identify a compatible and acceptable approach as well as to integrate control system requirements with normal mission operations. A significant part of any flight demonstration is evaluation of the control system utility and versatility.

The following items are suggested for supporting research and technology activity:

1) Simulations of control system and element alternatives;
2) Investigation of inner-loop force-feedback; and
3) Investigation of minimum acceptable TV requirements.

VII. ON-ORBIT SERVICER PRELIMINARY DESIGN

Fifteen on-orbit servicer concepts were screened and evaluated to select the pivoting arm concept as the most effective. This evaluation and the associated results are presented in sections B and C of chapter IV. The TRW design was selected to represent a group of four pivoting arm type servicers: RI UOP B (internal), MSFC, TRW, and Bell Aerospace cylindrical coordinate. The evaluation of all the on-orbit servicers, and especially the pivoting arm group, established in the project team a basic level of knowledge in the functional and mechanization aspects of on-orbit servicers. This "bank of knowledge" formed the basis for performing three subtasks which led to the final level of on-orbit servicer design knowledge. The first subtask was to formulate a comprehensive list on on-orbit servicer design requirements which are discussed in section A. The second subtask involved a top level investigation of several alternative pivoting arm servicer configurations. The objective was to utilize as much as possible the most effective features of all four of the pivoting arm designs which had been reviewed. The third subtask (section C) was to configure and fabricate a semi-functional model of a pivoting arm servicer with the associated stowage rack, SRUs and spacecraft. The very significant objective of this third subtask was to provide an effective and efficient "learning tool" for expanding our engineering design knowledge. By sequencing (manually) the pivoting arm through the series of operational steps involved in module exchange with a three-dimensional mockup, a realistic and focused discovery process took place.

This final level of on-orbit servicer design evaluation led to the pivoting arm on-orbit servicer preliminary design illustrated in Figure VII-1. The approach taken to arrive at this preliminary design has resulted in a design, not only very advanced for the preliminary stage, but also one that is effective and realistic because it has evolved through a systematic engineering process. The details of our pivoting arm on-orbit servicer are discussed in section D of this chapter and are related to engineering design drawings. Two significant conclusions have been drawn from this engineering design effort:

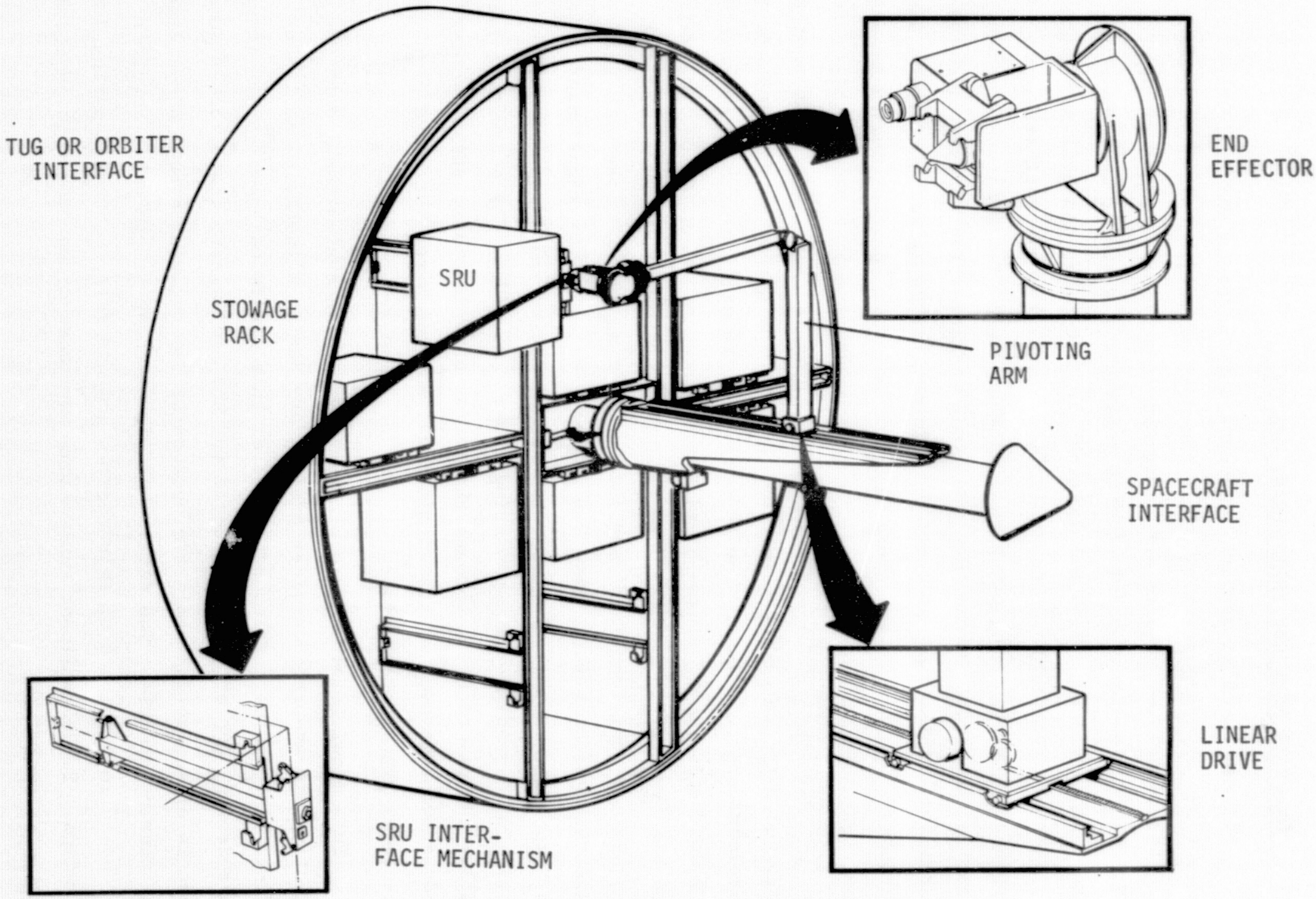


Figure VII-1 Pivoting Arm On-Orbit Servicer

1) Analysis, design, engineering test unit fabrication, and evaluation of on-orbit servicers should continue, and
2) No advancements in the state of the art are required for the pivoting arm on-orbit servicer.

A. ON-ORBIT SERVICER DESIGN REQUIREMENTS

A realistic and complete listing of requirements evolves naturally through an iterative process during a study which reviews and analyzes other concepts and then generates an improved design. The iterative process has taken place during this study and has resulted in the on-orbit servicer design requirements listed in Table VII-1. An important part of the generation of this requirements list has been the technical interchange with MSFC design engineers.

Some of the requirements in Table VII-1, like minimize degrees of freedom (resulting in minimum complexity and weight), are fundamental to most spacecraft equipment. These types of fundamental requirements are a natural integral part of our designer's approach to all space equipment. Requirements, like module mass range from 0 to 700 pounds, and the maximum module sizes, were arrived at by surveying the pertinent references to determine what maximum module weight and volume have been indicated. An accelerometer for the GRAVSAT spacecraft represents the heaviest module to be exchanged and thus sets the 700 pound upper limit.

Cost impacts interrelate as a forcing factor with many of the requirements for many reasons, e.g., cost to design and manufacture a servicer; cost to utilize a servicer in the STS program, and cost impacts on related equipment. Stowed length and weight are examples of requirements which can drive the spacecraft user's cost up when the launch cost reimbursement policy is based on them.

The requirement for the servicer mechanism to have a tip force greater than twenty pounds in worst configuration is based on good engineering practice for a space application of the servicer type. It is significant to note that the tip force requirement interrelates with the attach/latch actuator located in the end effector. The large forces required for making and breaking connectors are generated by the end effector actuator and do not impose forces on the servicer mechanism arm. The tip force level is

Table VII-1 On-Orbit Servicer Design Requirements

Minimize Degrees-of-Freedom
Module Mass Range 0 to 700 Pounds
To Handle and Stow Modules of the Following Size Characteristics:
Large - 40 X 40 X 40 Inches
Medium - 26 X 26 X 40 Inches
Small - 15 X 15 X 40 Inches
Minimize Stowed Length
Tip Force > 20 Pounds In Worst Configuration
Attach/Latch Actuator Located In End Effector
Time to Replace One Module - 10 Minutes
Generate Operational Status Signals
Minimize Sliding Friction Areas
Be Compatible With Orbiter/Tug/EOTS Electrical Power
Be Compatible With Automatic, Supervisory, and Remotely Manned Control
Satisfy All Latch/Attach Mechanism Guidelines
Compatible With Operations at Orbiter, Tug (IUS, FCT) EOTS
Compatible With Most Automated Spacecraft
Multiple Spacecraft Capability per Mission
Probability of Mission Success = 0.98
Reusable for 100 Missions
Lifetime of Five Years
Provide Failed Module Temporary Stowage
Provide Module Environmental Control (Thermal, Radiation, Contamination)
Operate Module Latches
Compatible With EVA
Compensate for Tolerances/Misalignments In 6 DOF
Withstand Orbiter Crash Loads
No Ability to Exchange Modules In One-G
Operable In One-G, No Modules
Lightweight

compatible with that for the SRMS and has been shown to be reasonable for module exchange in a Martin Marietta simulation.

Time to replace one module should be under ten minutes. This upper time bound has been arrived at from past Martin Marietta manipulator simulations and references in the literature.

Very important requirements are that the servicer be compatible with operations at the orbiter, tug (IUS, FCT), and EOTS, with most automated spacecraft as well as with the SEPS, and free-flying geostationary servicer. These two requirements are important from an economic standpoint and to establish spacecraft user acceptance.

Operation of the servicer in one-g without a module is needed to provide the capability of ground checkout both on a subsystem level after fabrication and at the launch facility.

B. PIVOTING ARM SERVICER ALTERNATIVE CONFIGURATIONS

The TRW pivoting arm on-orbit servicer was selected as discussed in Chapter IV as being the most effective of its group and of the fifteen servicers in the total field. The group consisted of: (1) the TRW pivoting arm, (2) MSFC pivoting arm in three evolutionary stages, (3) RI pivoting arm, and (4) Bell Aerospace Corporation cylindrical coordinate system. A top level formulation of alternative pivoting arm concepts was performed. The objective was to try to utilize many of the desirable features of the six existing designs. In this top level formulation, minimum as well as maximum changes from the six existing pivoting arm concepts were considered.

Since this study was begun, the TRW pivoting arm concept has evolved to that shown in Figure VII-2 where the linear motion is at the wrist. This results in a minimum stowage length of 105 in. and an operating length of 156 in. for the servicer mechanism and docking probe. A tertiary link has been added to provide some of the arm length and to orient the end effector to the module attach point. The extended arm length is 57 in. which is not long enough to reach the outer edge of a stowage rack (88 in.) that is the tug diameter.

The results of this investigation are shown in Figures VII-3, VII-4, VII-5, and VII-6. All four alternative concepts shorten the servicer mechanism operating distance which is defined as the distance from the front of the stowage rack to the front surface of the spacecraft. This is a very desirable objective because operating distance can be reflected in stowage length which in turn affects launch costs charged to the user. Alternative

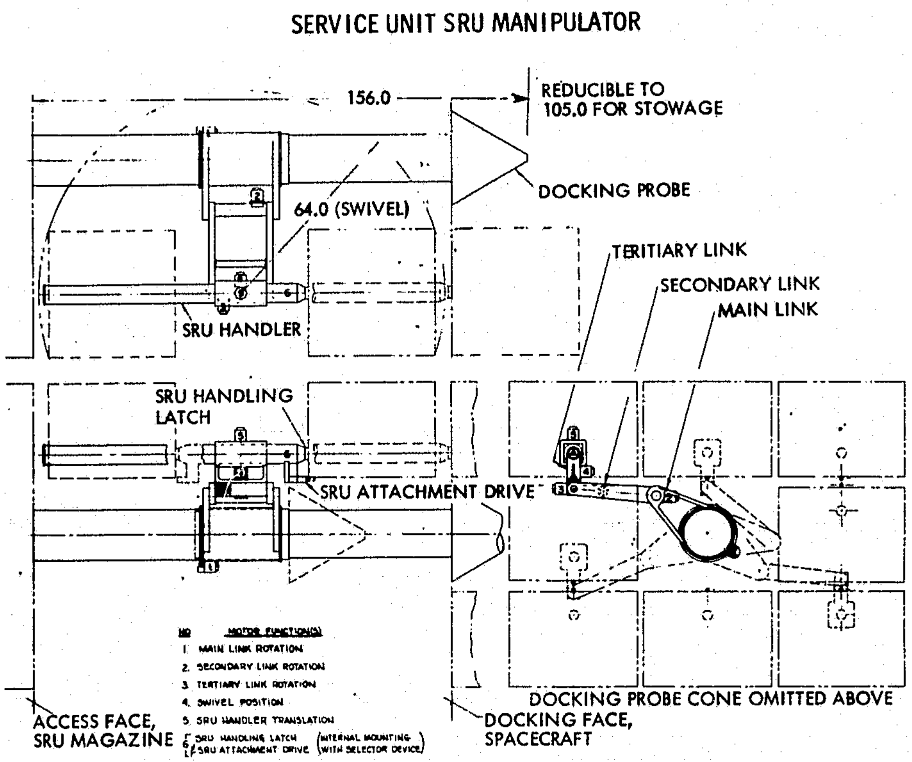


Figure VII-2 Current Configuration of the TRW Pivoting Arm

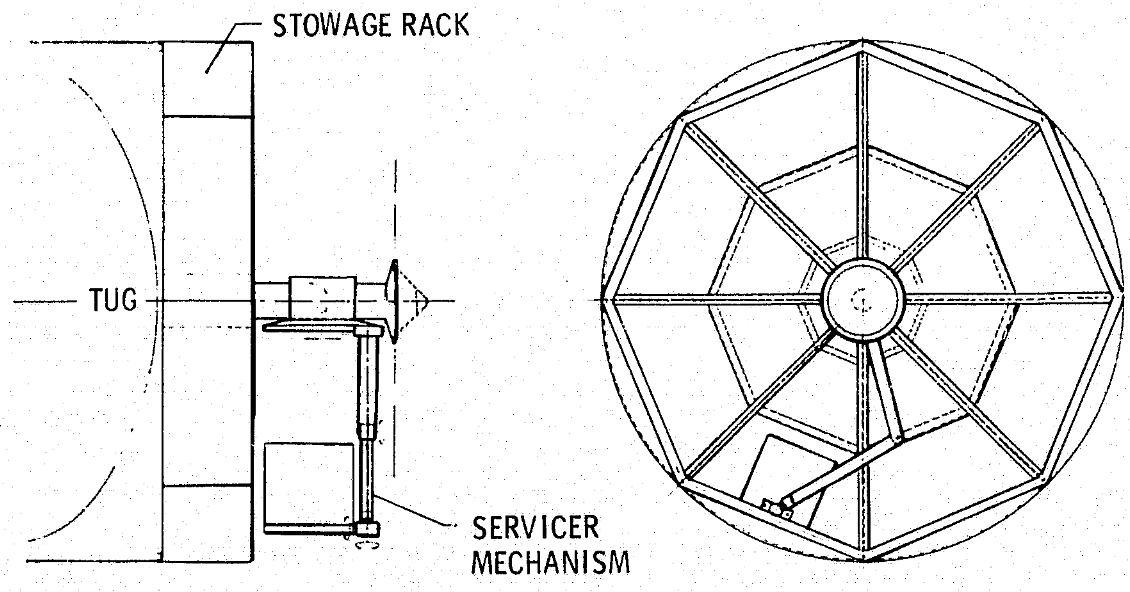


Figure VII-3 Pivoting Arm Servicer - Alternative A

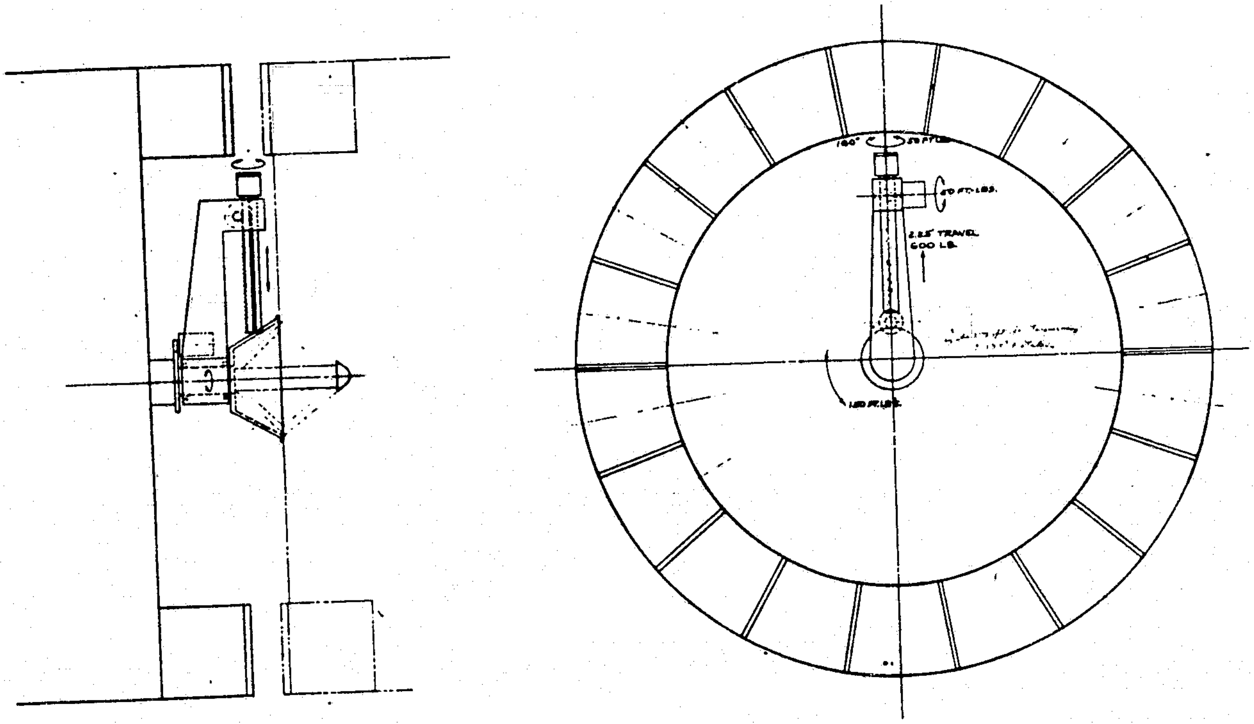


Figure VII-4 Pivoting Arm Servicer - Alternative B

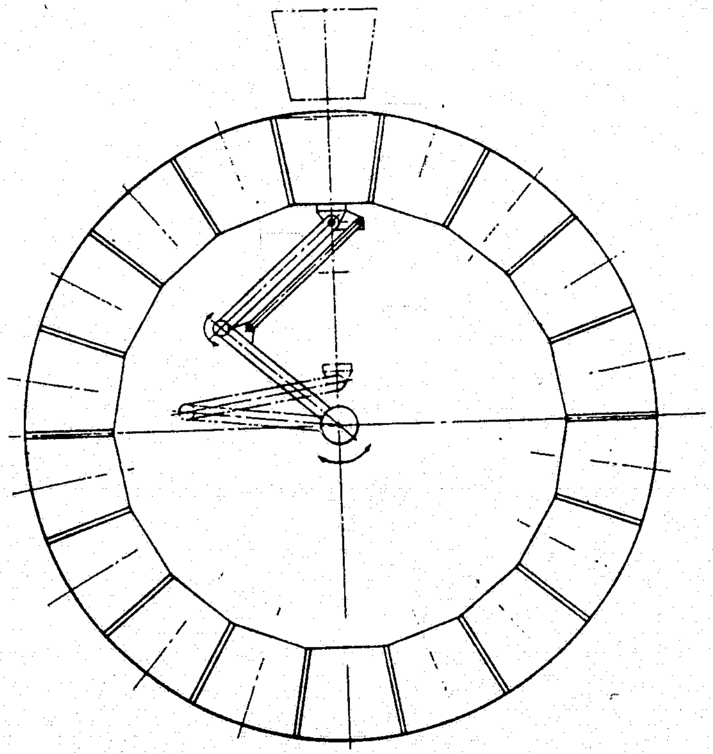


Figure VII-5 Pivoting Arm Servicer - Alternative C

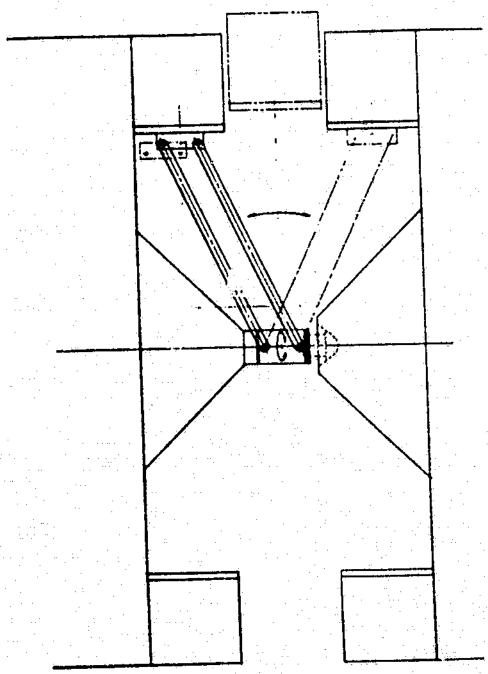


Figure VII-6 Pivoting Arm Servicer - Alternative D

concepts B, C and D represent almost a minimum operational distance for module exchange. They also are very simple mechanizations. However, this is done at the expense of losing a significant portion of the central region of the stowage rack and spacecraft for module location. Essentially, one tier of modules can be accommodated. Also, during the module exchange, the modules pass outside of the stowage rack/spacecraft envelope.

Out of the four alternatives, concept A was the only concept rated effective. It does not have limited module location, and thus it has excellent volumetric efficiency. Concept A shortens significantly the operational distance for module exchange from the TRW design because the linear drive has been transferred to the main central support structure. Thus, as a module is turned end for end, the linear drive can be driven to compensate for the turn around motion. The operating distance is thus shortened. The TRW design has the linear drive in the end effector and can't benefit from the same feature. Also, concept A can hinge the servicer mechanism to shorten the stowed distance. Alternative concept A was selected to be carried forward in the detailed design effort.

C. PIVOTING ARM OPERATIONAL AND INTERFACE INVESTIGATION

The generation of our on-orbiter servicer design, discussed later, was aided significantly by an operational and interface investigation which centered around a pivoting arm servicer and spacecraft mockup. Figure VII-7 shows all of the total system parts which were mocked up for the investigation. The parts mocked up reflect the design parameters which were variables in the investigation. These variables included:

1) Side- and bottom-mounted module interface mechanisms,
2) End effector for side- and bottom-mounted modules,
3) Different spacecraft: INTELSAT and Large X-ray Telescope,
4) Stowage rack and spacecraft with side- and bottom-mounted modules,
5) Different module sizes (making largest module removable), and
6) Stowed and operating orientations of the pivoting arm.

The stowage rack was made to accommodate side and bottom mount interface mechanisms by fabricating the stowage rack so that one half could accept bottom-mounting modules and the other half could accept side-mounting modules. Plexiglass was used for the outer structure of the stowage rack

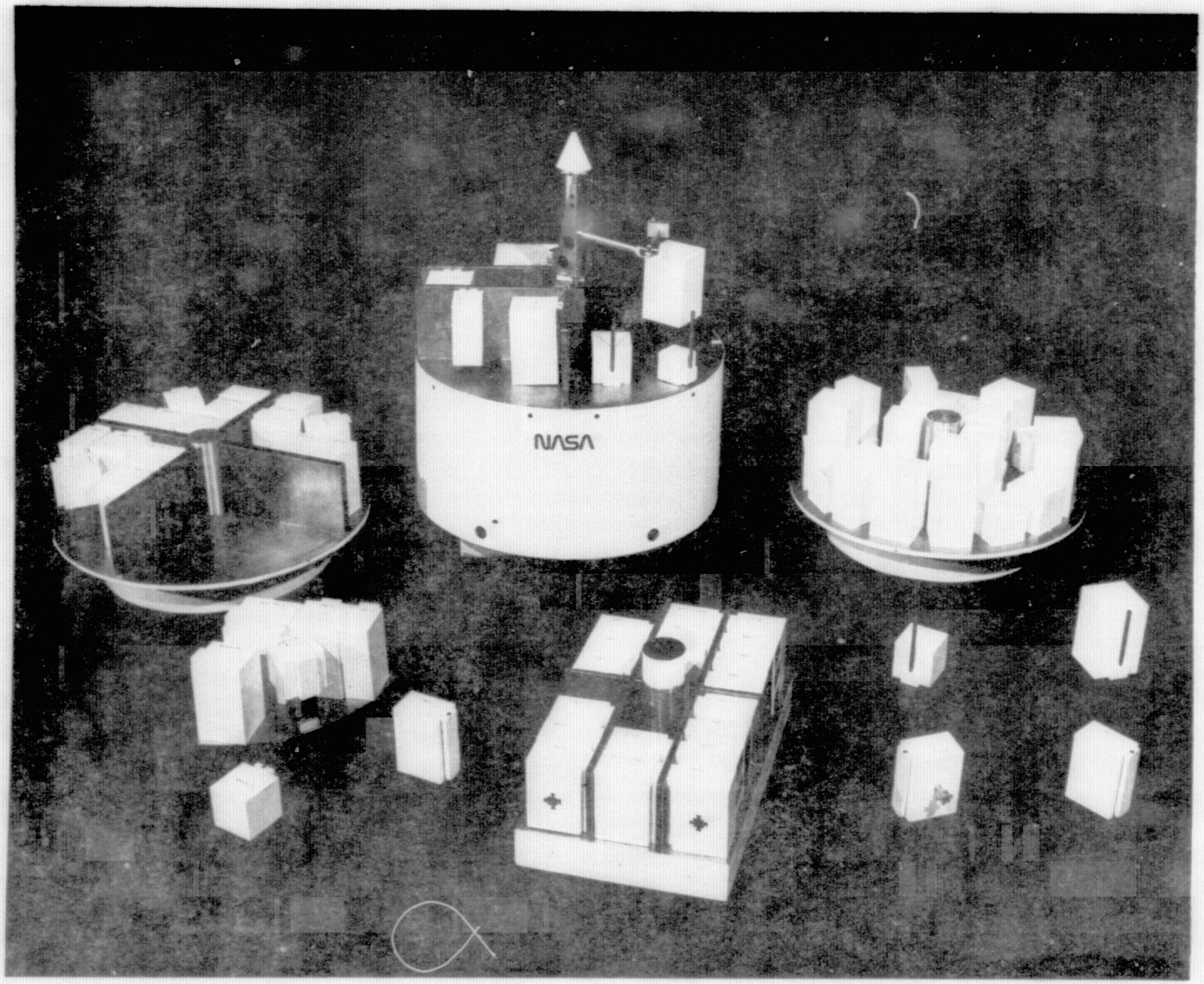


Figure VII-7 Pivoting Arm Servicer Mockup

and the module exchange region of the spacecraft to aid in observing module interference with other modules and the spacecraft structure during module exchange.

Figure VII-8 is an artist's illustration of a tug performing a docking approach with an INTELSAT in high earth orbit. The pivoting arm servicer is mounted on the front of the tug. The docking probe is extended and the servicer mechanism is in operational status but positioned back against the stowage rack front face to minimize potential hazards during the docking operation. Figure VII-9 is a photo showing the same servicing operation as it was investigated with the mockup.

Servicing of a large x-ray telescope with the pivoting arm servicer at the orbiter is illustrated in Figure VII-10. The stowage rack is shown in the operational configuration where it is supported from side mounts extending up from the cargo bay longerons. The docking probe which is an integral part of the servicer is extended and ready for docking. The stowage rack is moved from the stowed launch location to the operational location shown by the SRMS. Figure VII-11 is a photo showing the same servicing operation as it was investigated with the mockup. However, the cargo bay was not mocked up and potential operational interferences with the sides of the cargo bay had to be visualized.

The mockup investigations of pivoting arm servicing for orbiter or tug applications evidence how readily this type of on-orbit servicer can be adapted to servicing of both low and high earth orbit spacecraft. The two half sections of the stowage rack were examined for ease of module installation. The side-mounting half, with the vertical webs, appeared to be more restrictive in module locations. The bottom-mounting half, with no dividers, appeared to be more flexible in module location. However, both halves were easily able to stow a representative half complement of modules. This was even true for the multiple spacecraft servicing situation. It was also relatively easy to locate the large (40 in. cube) modules for either interface mechanism form.

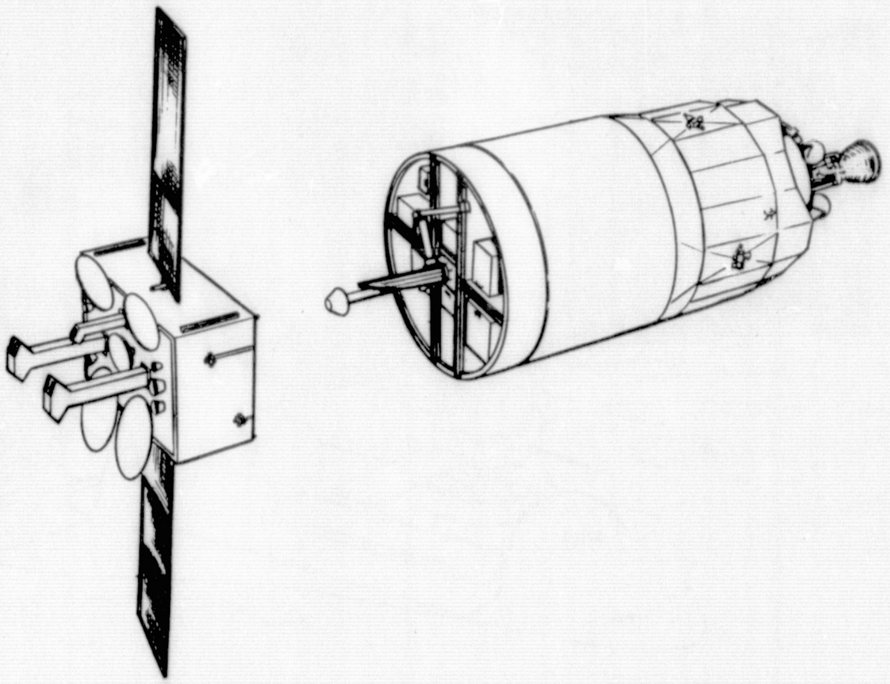


Figure VII-8 Servicing the INTELSAT via the Full-Capability Tug

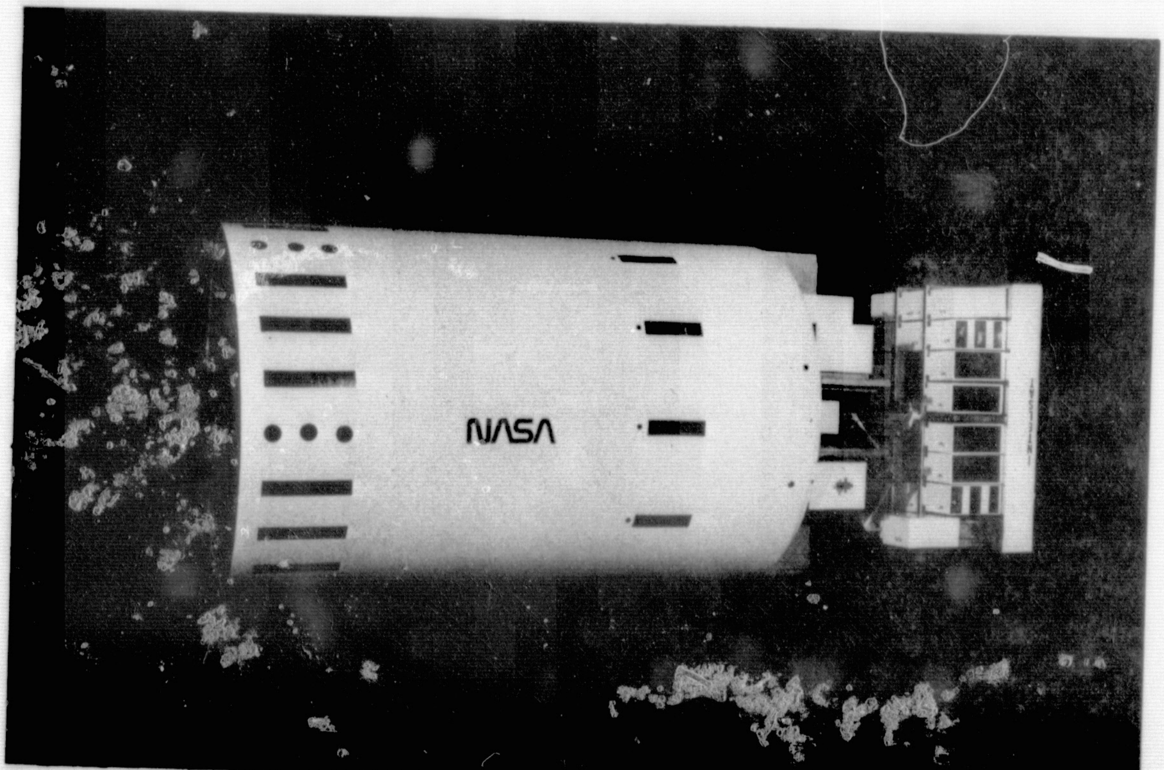


Figure VII-9 Details of the Pivoting Arm Servicing the INTELSAT

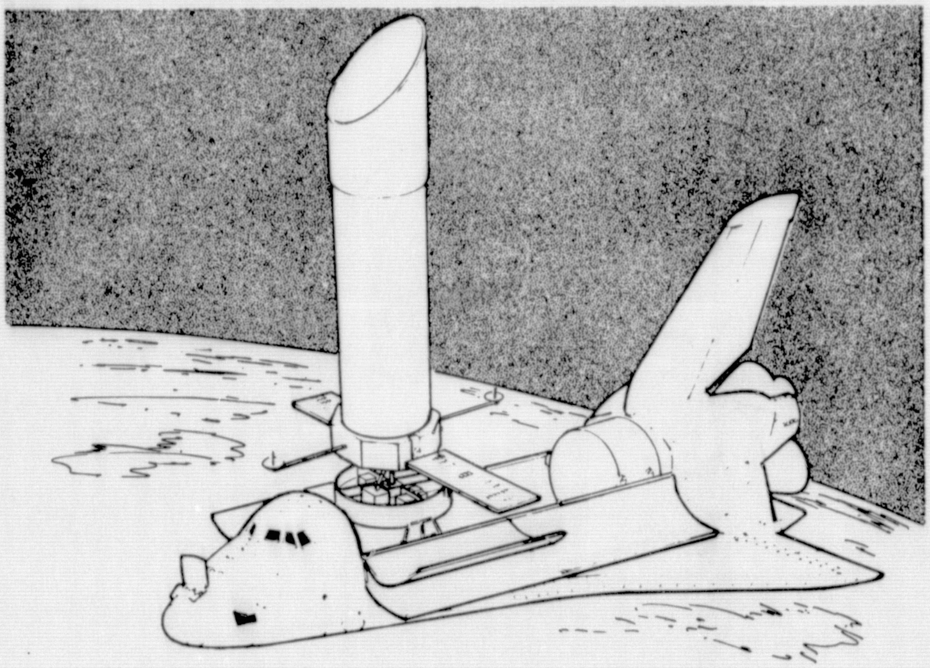


Figure VII-10 Servicing the Large X-Ray Telescope at the Orbiter

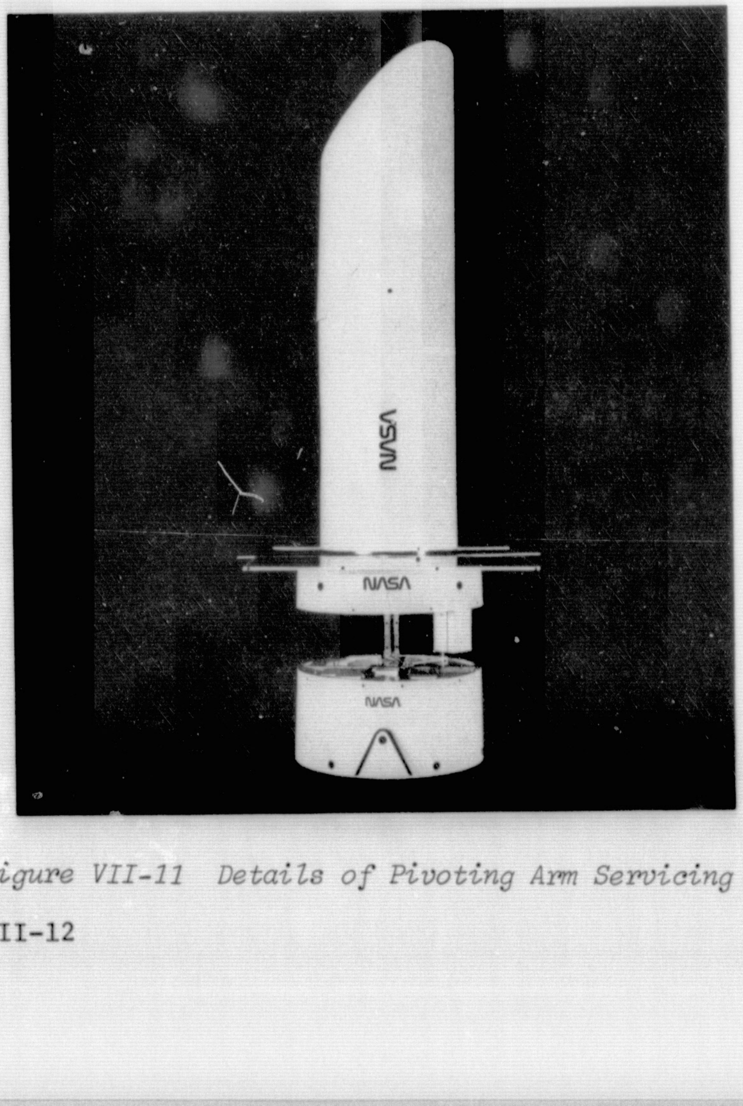


Figure VII-11 Details of Pivoting Arm Servicing the Large X-Ray Telescope

The LXRT was configured for both bottom- and side-mounting of modules. The resulting configuration appeared different. Bottom-mounting arranged the modules radially, while the side-mounting arranged the modules generally rectangularly. In both cases, all the required modules could be installed with some excess space.

A single end effector design was derived which was adaptable to both versions of the SRU interface mechanism. The attach interface, between the end effector and the baseplate, occurs at the exterior spacecraft surface for the side-mounting case, and about 9 inches from the mounting surface in the other case. This amounts to a 31 inch difference in length of the end effector. The approach suggested is to move the attach interface up from the bottom so that it occurs in the same plane for both bottom- and side- mounting SRU interface mechanisms.

A typical operational module exchange sequence is shown in the series of photos of Figure VII-12. As the pivoting arm is manually moved through the operational steps, module trajectory paths and interference clearances were observed. The mockup provided an excellent three-dimensional investigation. No operational or interference problems were noted during the investigation. However, utilization of the mockup for design investigations was only really just started. Different parts of the mockup can be changed to allow much more detailed investigations of design variables in the future.

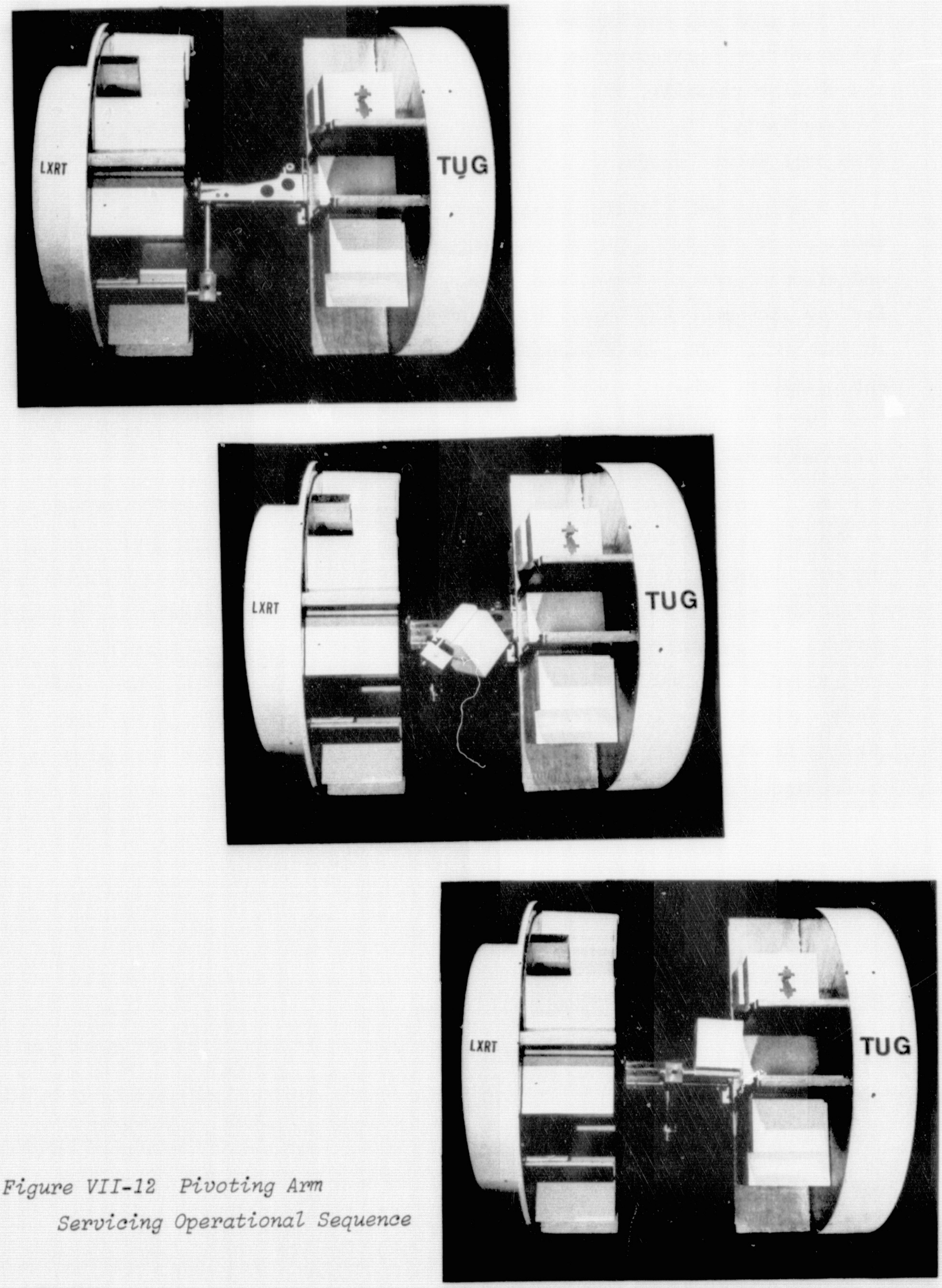


Figure VII-12 Pivoting Arm Servicing Operational Sequence

D. ON-ORBIT SERVICER PRELIMINARY DESIGN

During the course of this study and as discussed above, it was determined that for spacecraft servicing (module exchange), the pivoting arm device would be the most useful and the simplest mechanism. A more detailed conceptual design of this mechanism has been completed and is presented here.

This design has only two major components: (1) a pivoting arm servicer mechanism (as is shown in Figure VII-1), and (2) a stowage rack for module transport. A docking mechanism is also shown for reference and so that the mechanical interface aspects can be more easily visualized. The servicer mechanism and the stowage rack were designed separately with interfaces for individual removal and replacement. This allows for simple removal for maintenance and also for quick ground reconfiguration. Stowage racks can be configured and loaded for particular flights prior to attachment to the carrier vehicle. It may be desirable to have available several stowage racks for this purpose. The docking mechanism and stowage racks are structured to accept longitudinal and bending loads imposed by the spacecraft during the docking and servicing phases. This structure is also capable of handling launch and crash loads when in the servicing only configuration. The concept is also adaptable to mounting a spacecraft on the docking mechanism during launch and reentry. For this case, the structure is capable of handling fore and aft launch and crash loads but will require auxiliary side load supports when the spacecraft is in the orbiter cargo bay.

This concept does not preclude the use of a peripheral docking mechanism as proposed by McDonnell Douglas Astronautics Corporation. The central docking device is shown here for the purpose of showing the advantage of a central docking mechanism in regard to interfacing with variable spacecraft diameters and the removal of interferences at the outer edge of the stowage rack.

1. Pivoting Arm Servicer Mechanism

The pivoting arm consists of three primary sections, the central positioning mechanism, the arm mechanism, and the end effector as shown in

Figure VII-13. The central positioning mechanism incorporates two of the servicer mechanism's four degrees of freedom--shoulder roll and linear travel. The arm mechanism consists of an inner arm, an elbow, an outer arm, and a wrist. It incorporates the elbow roll and wrist roll degrees of freedom as well as the two position pitch drive. The end effector provides the module attachment interface and contains the motor to drive the SRU interface mechanism latches. The end effector and the SRU interface mechanisms are described in Chapter V. The figure shows the travel of the various elements and the location of the edge of the spacecraft during module exchange. The docking probe is shown extended. After docking, it is retracted so the spacecraft takes the position shown.

The central positioning mechanism provides the first two degrees of motion--shoulder roll and linear travel--, is the interface with the arm stowage mechanism, provides space for the docking probe, and interfaces with the arm mechanism. The physical expression of these functions is shown in Figure VII-14. The figure also shows the details of the arm stowage mechanism which is described below. The interface between the shoulder roll drive and the stowage rack is a simple hinge and latch arrangement. These parts are well spaced to provide good bending and torsional stiffness as well as good alignment accuracy. The shoulder roll drive permits positioning the end effector along any radius of the stowage rack, or spacecraft. Precision and stiffness are obtained by use of two large diameter, thin section bearings. Note that the inner bearing race support structure (center post) also forms the support structure for the docking probe. This probe can be installed or not with no effect on servicer mechanism operation. The roll drive uses a DC torque motor with a 51:1 gear reduction, most of which is obtained from a pinion through an idler to a large external gear fastened to the center post. The roll drive motor, gearing, brake, tachometer generator, and potentiometer are fastened to the linear drive support and move around the center post. Backlash can be controlled at the output gear mesh by shimming the roll drive housing or mounting the idler gear in eccentric bushings to obtain the desired low level.

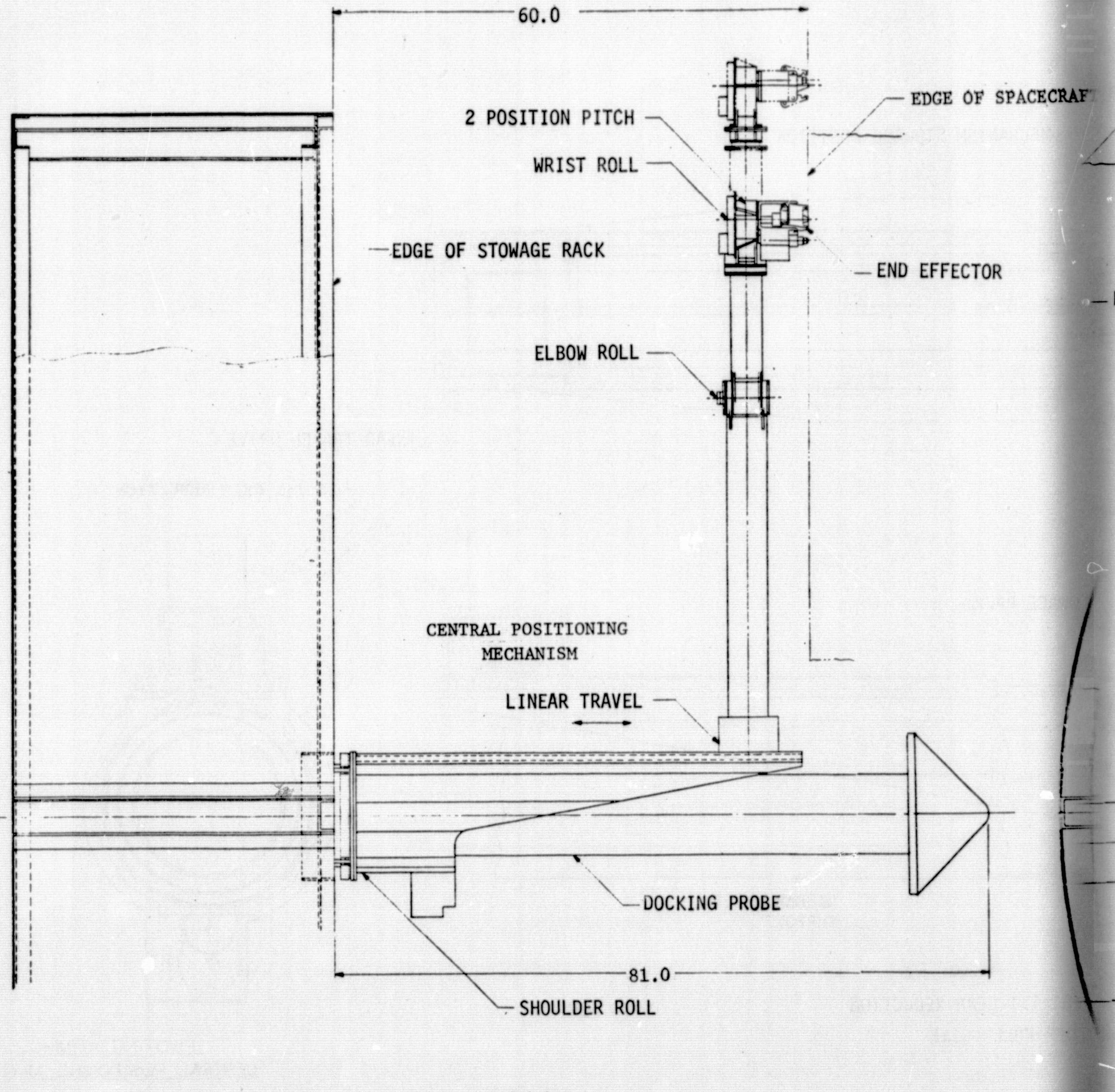


Figure VII-13 Pivoting Arm Servicer Mechanism

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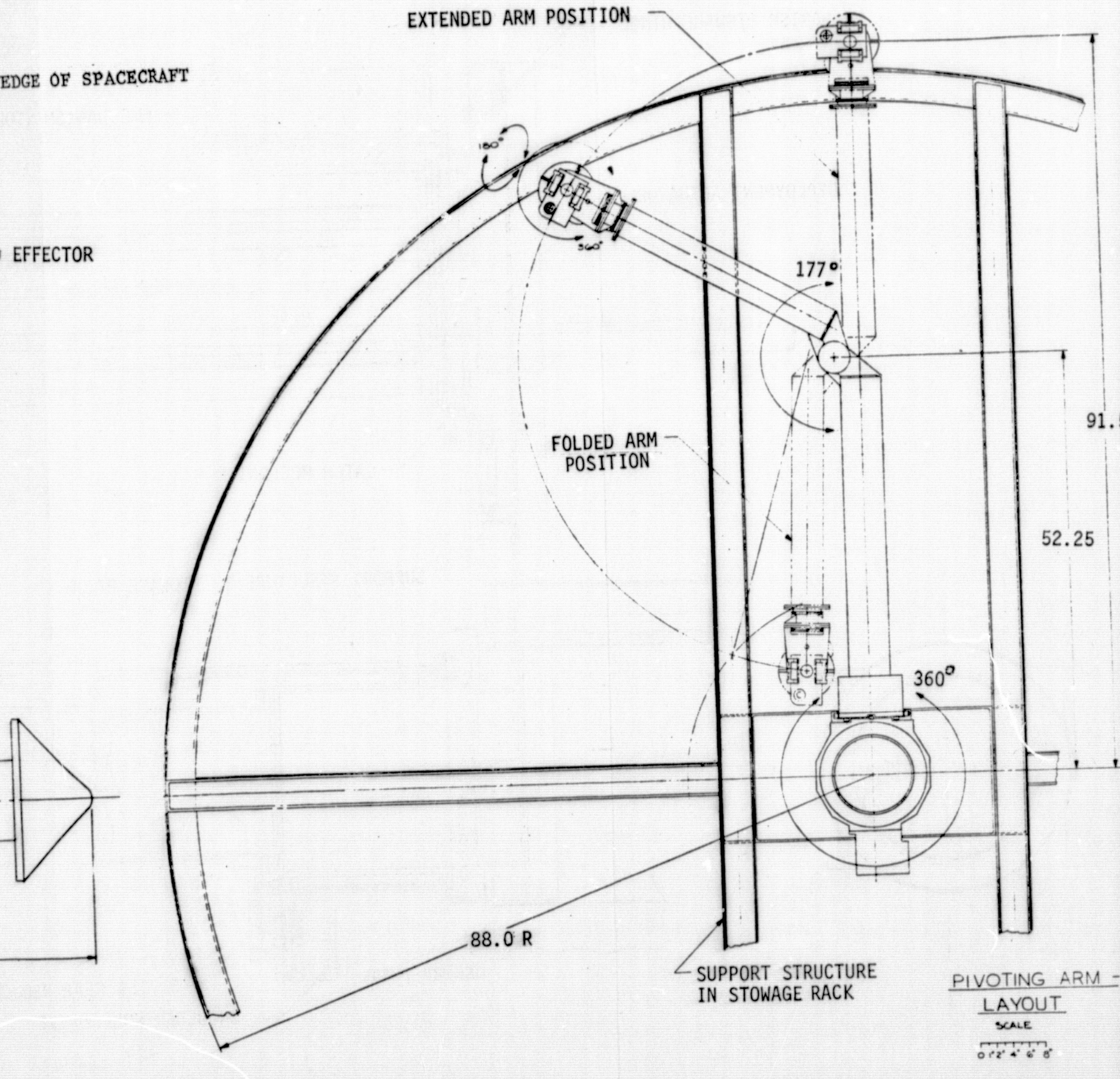

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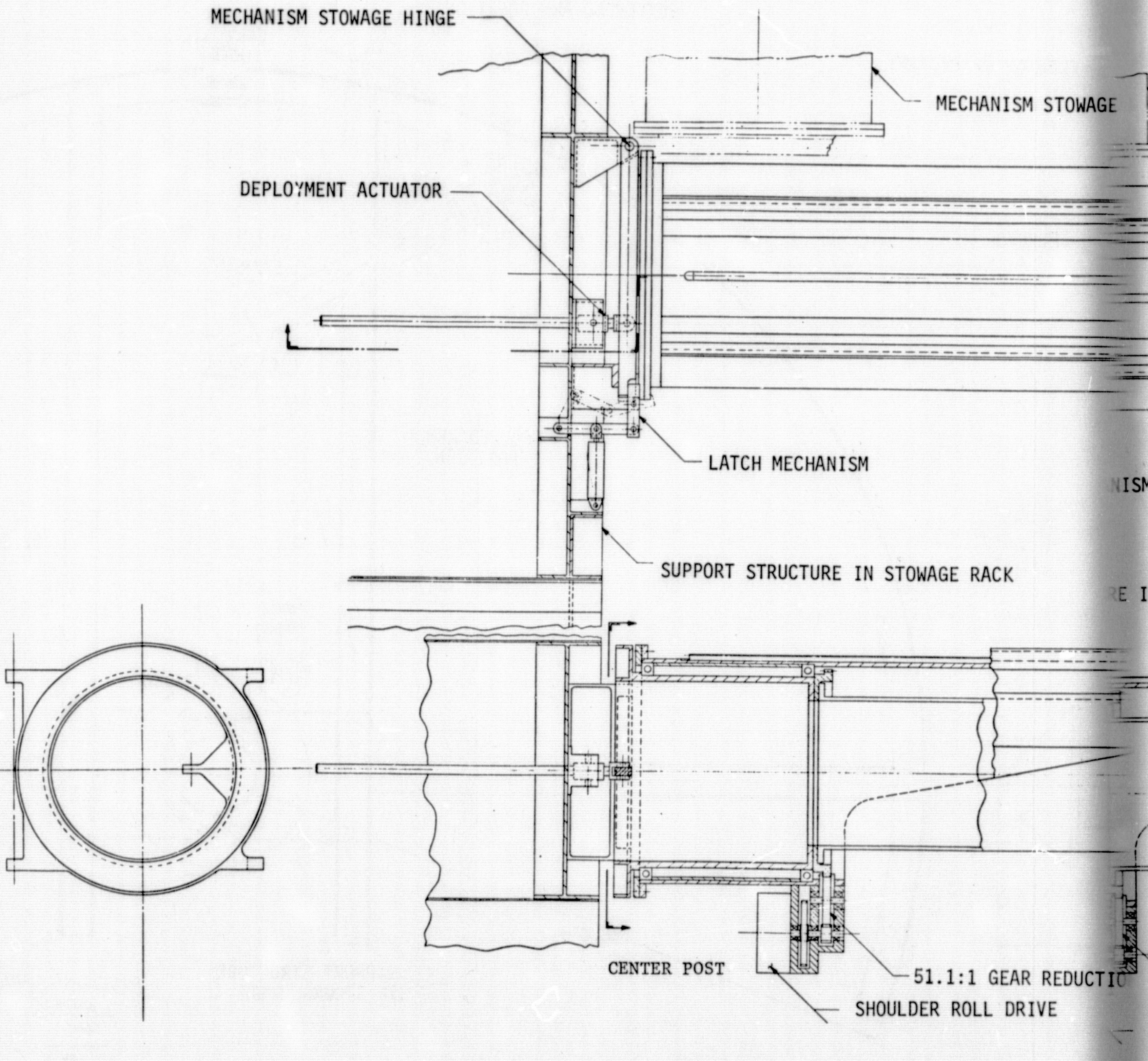

Figure VII-14 Pivoting Arm - Central Positioning Mechanism

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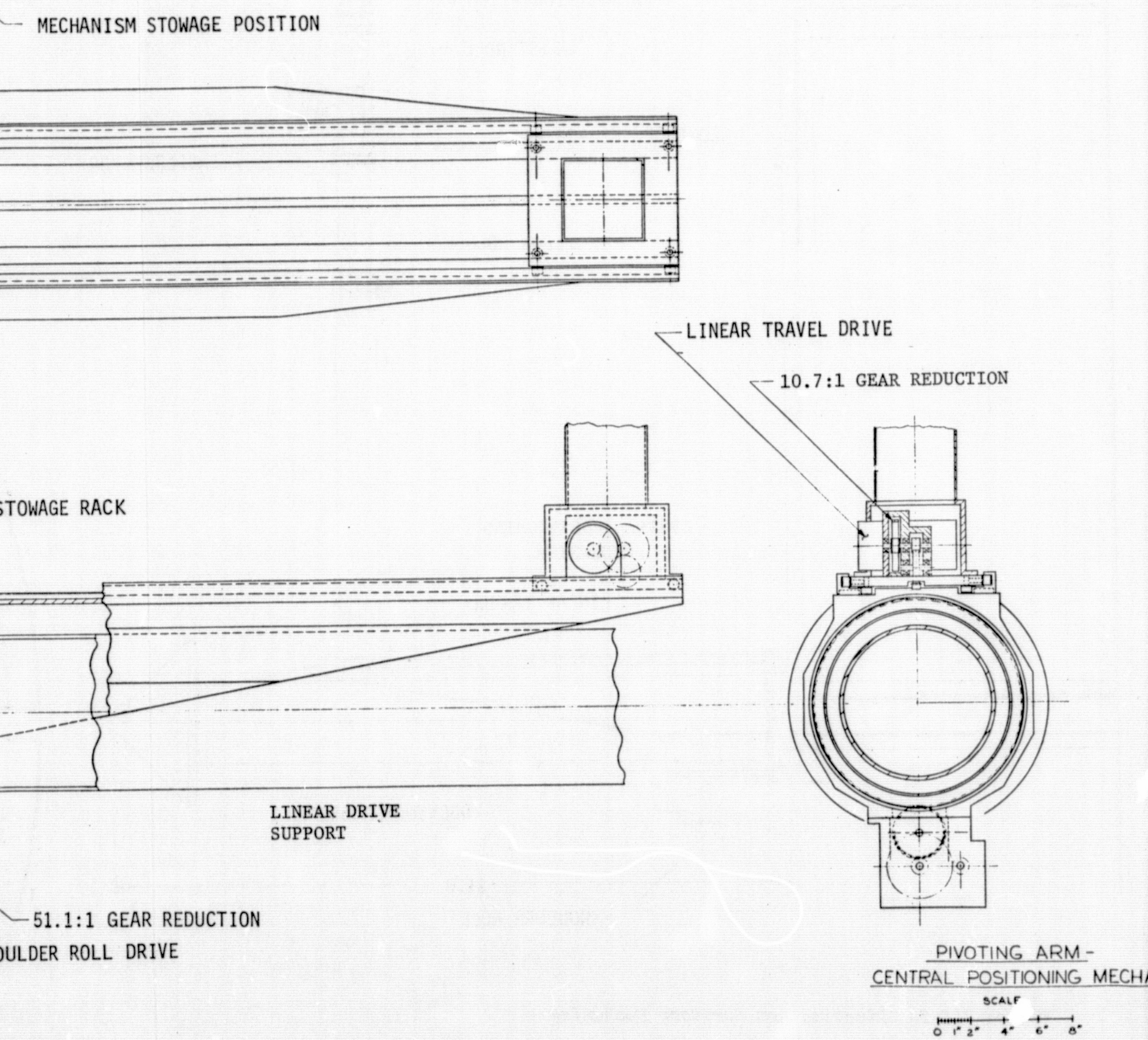

Mechanism

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The space between the center post and linear drive support can be used for wire bundle motion. Flat cable, or flat wiring, in the form of a loop could be put into this space, or an appropriate enlargement.

A linear drive was selected as the simplest method of obtaining the desired motion parallel to the stowage rack/spacecraft centerline (X motion). Linear drives tend to be heavier and more complex than rotary actuators and are harder to configure for good stiffness and alignment. However, no reasonable form of rotary actuator configuration was discovered in our preliminary layouts. All the pivoting arm forms in the literature also used at least one linear drive.

The linear drive support provides the required torsional and bending stiffness to support the arm mechanism. These structures are all designed to give the desired structural natural frequency when the maximum mass module is supported in the softest arm configuration. The 20 lb tip force should not cause too large a deflection of the arm tip. The drive support has been sized to provide rigid local support of the linear drive rollers and so that the structural deflections due to pinion and rack loadings will be acceptable.

The linear drive has a rack and pinion as final output gearing. They are located midway between the locating rollers to minimize deflections and backlash. As there is no magnifying effect in the rack and pinion linear drive (as is the case between a rotary actuator output gear and wrist motion), backlash is not a problem. Four rollers are used for Z and pitch alignment and four other rollers are used for Y, roll, and yaw alignment. While three rollers are theoretically all that are required, four are used to obtain a slightly smaller package. The rollers and supporting tracks are sufficiently spaced to prevent binding.

The linear drive contains a DC torque motor, 10.7:1 gear reduction, brake, tachometer and potentiometer. Note that an idler is provided to connect the output pinion to the rack. The outer part of the drive provides the support and attachment for the inner arm. A sheet metal trough can be mounted to one (or both) side(s) of the roller guides to support the wire bundle loops. Again, flat cables or flat wiring can be used. By

attaching one end of the cable bundle near the mid point of travel, the unsupported length can be minimized.

The arm mechanism, with its associated elements is shown in Figure VII-13 in three different positions--extended, maximum operating reach, and folded. Unequal arm segment lengths (inner 52.25 in., and outer 39.25) were used to obtain the desired arm stowage characteristics. The extended arm dimension was chosen so that at 90 percent of full reach (maximum operating reach) the arm would reach any SRU interface mechanisms at the outer edge of the stowage rack. The minimum operating reach (25 percent of full reach) is compatible with any anticipated central location of the SRU interface mechanisms. The 25 and 90 percent figures are those typically used in manipulator design.

Figure VII-15 shows the arm mechanism details. The inner and outer arm segments are thin-wall, square aluminum tubes to provide the maximum cross sectional moment of inertia for the volume available. It was decided to design a four DOF arm. This means that there can be tolerance buildups in the two degrees of freedom that are not available for control as is discussed in Chapter VI. The effect here is that the arm segments must be designed to be soft enough to compensate for the tolerances yet stiff enough to be controllable in the other degrees of freedom. The selection of the required stiffness level is beyond the scope of the present effort.

The elbow drive uses two large-diameter thin-section well-spaced output bearings to provide the desired stiffness with light weight. The gear reduction is an internal gear, double pinion system that we have used successfully in several manipulator arms. It permits adjusting the backlash down to the low desired levels. The elbow drive contains a DC torque motor, brake, tachometer, and potentiometer along with the 110:1 gear reduction.

The wire loop bundle can be routed around the outside of the drive or placed in sheet metal cans at one or both ends of the drive. Again flat cable or flat wiring can be used. The limited travel and fewer wires simplifies the wire routing at this joint.

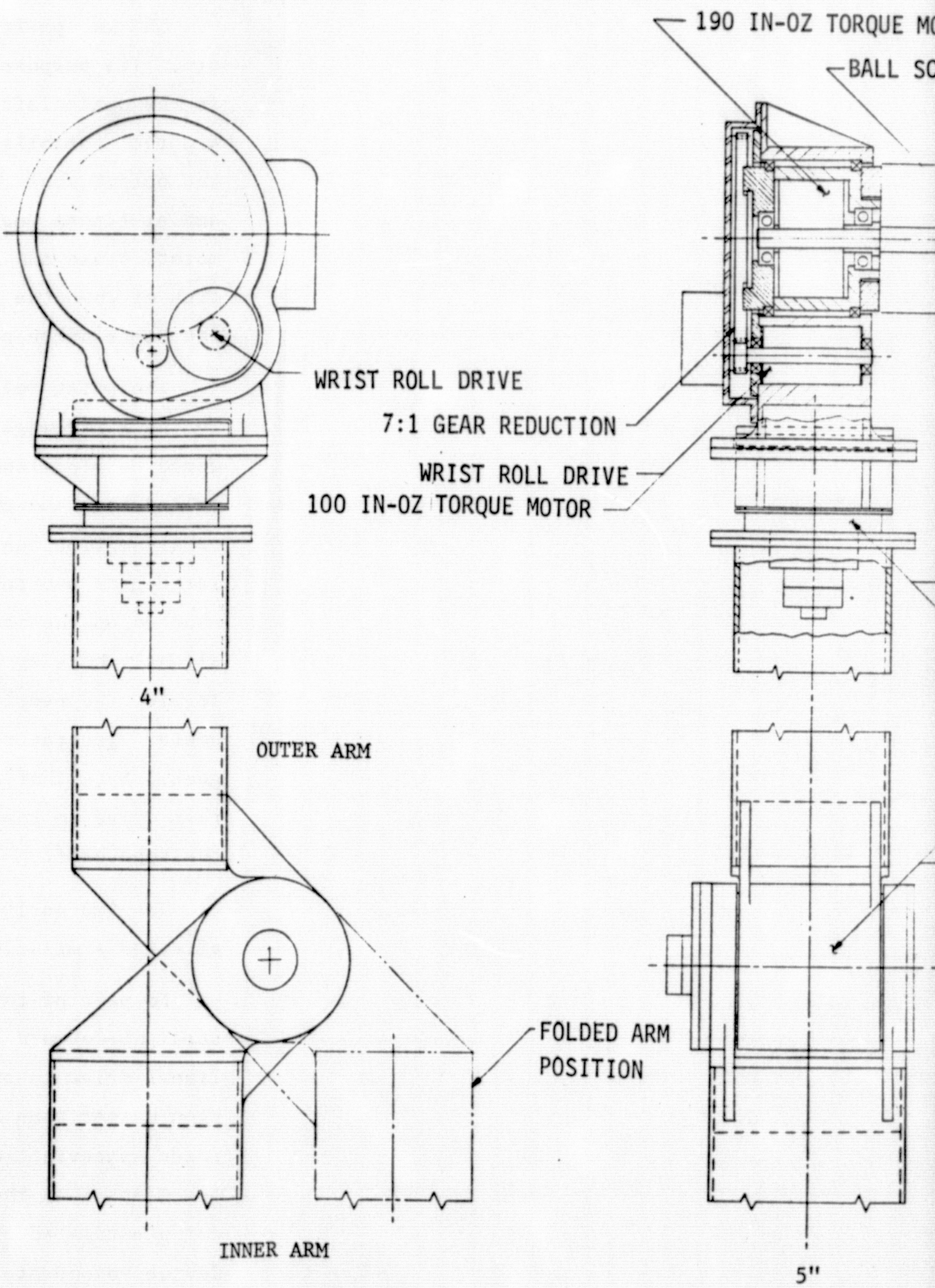
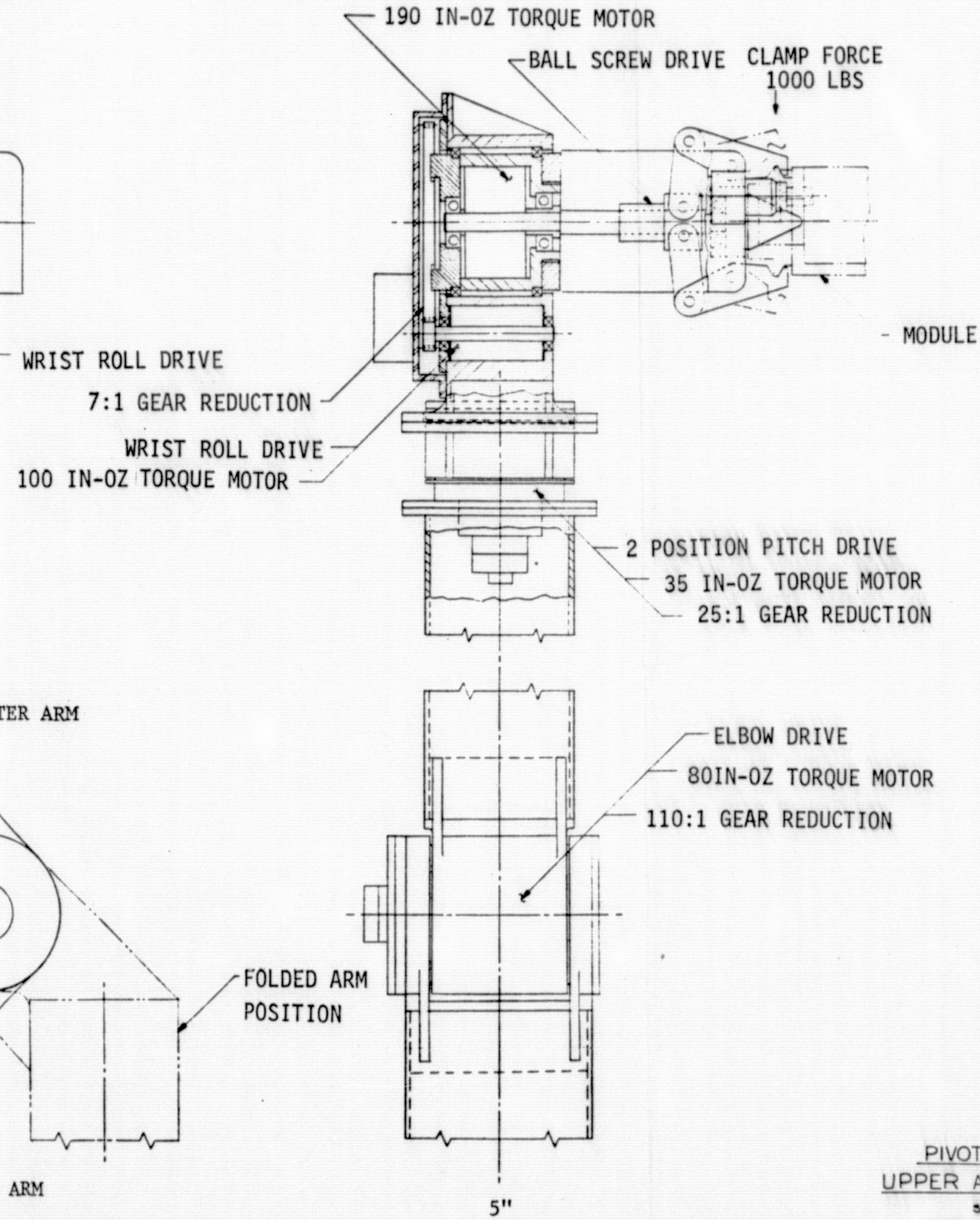


Figure VII-15 Arm Mechanism



PIVOTING ARM -
UPPER ARM MECHANISM

SCALE

0 1" 2" 3" 4"

Mechanism

FOLDOUT FRAME 2

The two position pitch drive is located at the wrist end of the outer arm. Its purpose is to turn the modules end for end so they may be placed in the spacecraft or stowage rack. The drive is thus an indexing and not a servo controlled drive. As the index positions will be hard stops at the output shaft, gearing backlash is not a problem. Good bearing accuracy and stiffness must be obtained. A 25:1 gear ratio is used along with the motor, brake and a potentiometer. The potentiometer provides an indication of when the joint is at a stop and provides a basis for generating the other drive signals as the module is turned end for end.

The wrist roll drive design was driven by the desire to minimize length in the X (stowage rack centerline) direction so that the operating length would be minimized. The outer form of the end effector (Chapter V) was cylindrical and thus could be readily mounted in large diameter, small cross-section bearings as is desired. The drive then took the form of a large gear mounted on the end effector and driven directly by a pinion mounted on the motor shaft. This provided a 7:1 gear ratio and meant a slightly heavier motor than usual. As the wrist requires the lowest torque levels, the penalty is acceptable. The other drive elements--brake, tachometer generator, and potentiometer--are geared directly to the large gear. Note that a full 360 degrees of travel are provided in the wrist roll drive so that any module can be positioned in any roll orientation on the spacecraft.

The two position pitch drive and wrist roll drive wire routing can be effected similarly to the methods suggested for the other rotary actuators.

No part of this design is particularly unique. It accomplishes a straight-forward task by well known means. We have limited the use of the linear drive to one operation. The use of the linear drive at this position rather than at the end effector allows for the shortest distance between spacecraft and stowage rack. The simple geometry of the pivoting arm means that the control system will be correspondingly simple. Whether doing automated or man-controlled functions, it is an exceedingly simple device to operate.

The pivoting arm servicer mechanism exchanges modules between the spacecraft and the stowage rack. The longer the separation distance between the face of the stowage rack and the spacecraft interface, the greater the weight required to maintain adequate structural rigidity to meet the module exchange positional accuracies. The operational separation distance required by our design shown in Figure VII-13 is 60 inches. This is a significant reduction from the TRW pivoting arm design which requires 105 inches for exchanging modules. Our design can exchange 40-inch modules in an operating length of 60 inches because the linear drive is on the main support structure at the shoulder rather than in the wrist as in the TRW design. Programming of the linear drive is required to drive it in a compensating direction as a module is rotated 180 deg during a module exchange operation.

A weight statement for the on-orbit servicer is given in Table VII-2 for the design described above. The data for the pivoting arm mechanism and for the stowage rack are given separately. The mechanism weight is close to what was previously estimated for the mechanism when the addition of the folding capability is taken into account. Note that the docking probe weight is not given.

The stowage rack weight is somewhat heavier than earlier estimates due to two factors. The weight of twelve tracks (baseplate receptacles) has been included for the first time. More importantly, the structure has been designed to take the launch and crash loads when a spacecraft is mounted on the docking probe. This additional requirement, which may not be valid, has contributed in large part to the weight increase.

Table VII-2 On-Orbit Servicer Weight Statement

PIVOTING ARM ASSEMBLY		136 lb
ROTATING TUBE	40	
UPPER ARM	5	
LOWER ARM	5	
ELBOW DRIVE	5	
LINEAR DRIVE	8	
HINGE PLATE AND BRACKETS	25	
GEAR	6	
BEARING SHAFT	15	
ARM BASE PLATE	5	
WIRING AND CONNECTORS	10	
END EFFECTOR	12	
STOWAGE RACK ASSEMBLY (EXCLUDING MODULES)		484 lb
OUTER SKIN (0.050)	110	
END SKIN (0.040)	98	
FRONT MEMBER	25	
BIG FRAME	55	
SHEAR PANELS (0.032)	70	
BACK ANGLES	27	
FITTINGS	41	
ANGLES	10	
INTERFACE MECHANISM TRACKS (12 AT 4 lb EACH)	48	
CONTROL ELECTRONICS ASSEMBLY		30 lb
PIVOTING ARM SERVICER TOTAL WEIGHT		650 lb

2. Stowage Rack

The stowage rack serves the following purposes:

1) Serve as support structure for modules, side-mount and bottom-mount,
2) May be designed as environmental enclosure and system support device for transported modules,
3) Provides carry-through structure for docking probe and spacecraft,
4) Acts as adapter to tug
5) Provides attachment location for airborne support equipment for mounting in orbiter

6) Provides support structure for docking system latches

7) Provides mounting structure for pivoting arm mechanism stowage device and its latches.

The stowage rack as shown in Figure VII-13 is a continuation of the 176 in. diameter tug outer skin. There are fore and aft ring frames, outer skin and suitable skin stiffeners. All crossing members are shown as shear panels and are used to support the module track assemblies which receive the removable modules. The front edge of the panel is supported by cross beams. These beams are required to resist side loading of 4 g's during crash conditions. They also serve as the front support member for attachment of the module track assemblies. At the rack center is a mechanical splice fitting to which all structural members are attached and to which the pivoting arm assembly attaches. Longerons are located at the outer skin to help spread out the longitudinal loads for equal distribution into the tug outer skin.

3. Arm Stowage Mechanism

An arm storage mechanism is located at the center of the stowage rack for folding of the total pivoting arm during nonoperating periods. This shortens the stowage length in the orbiter cargo bay. Servicer mechanism stowed length is a critical design parameter because of both structural and launch cost impacts. The TRW pivoting arm servicer design has a stowed length of 105 inches. In our study considerable design effort was focused on minimizing the servicer mechanism stowage length. Our design requires 21 inches for stowing the servicer mechanism. The details of the stowed configuration are shown in Figure VII-16. The small stowage length is arrived at by hinging the pivoting arm center post and the docking mechanism tube at the forward face of the stowage rack. In the stowed configuration, the pivoting arm then lies up against the face of the stowage rack. Note how the docking mechanism also lies flat against the front surface of the stowage rack. During the servicing operation the servicer mechanism would be rotated 90 deg to a position normal to the face of the stowage rack.

The lower and upper segments of the pivoting arm are stowed by driving the elbow roll to a position where the arm is folded completely back on

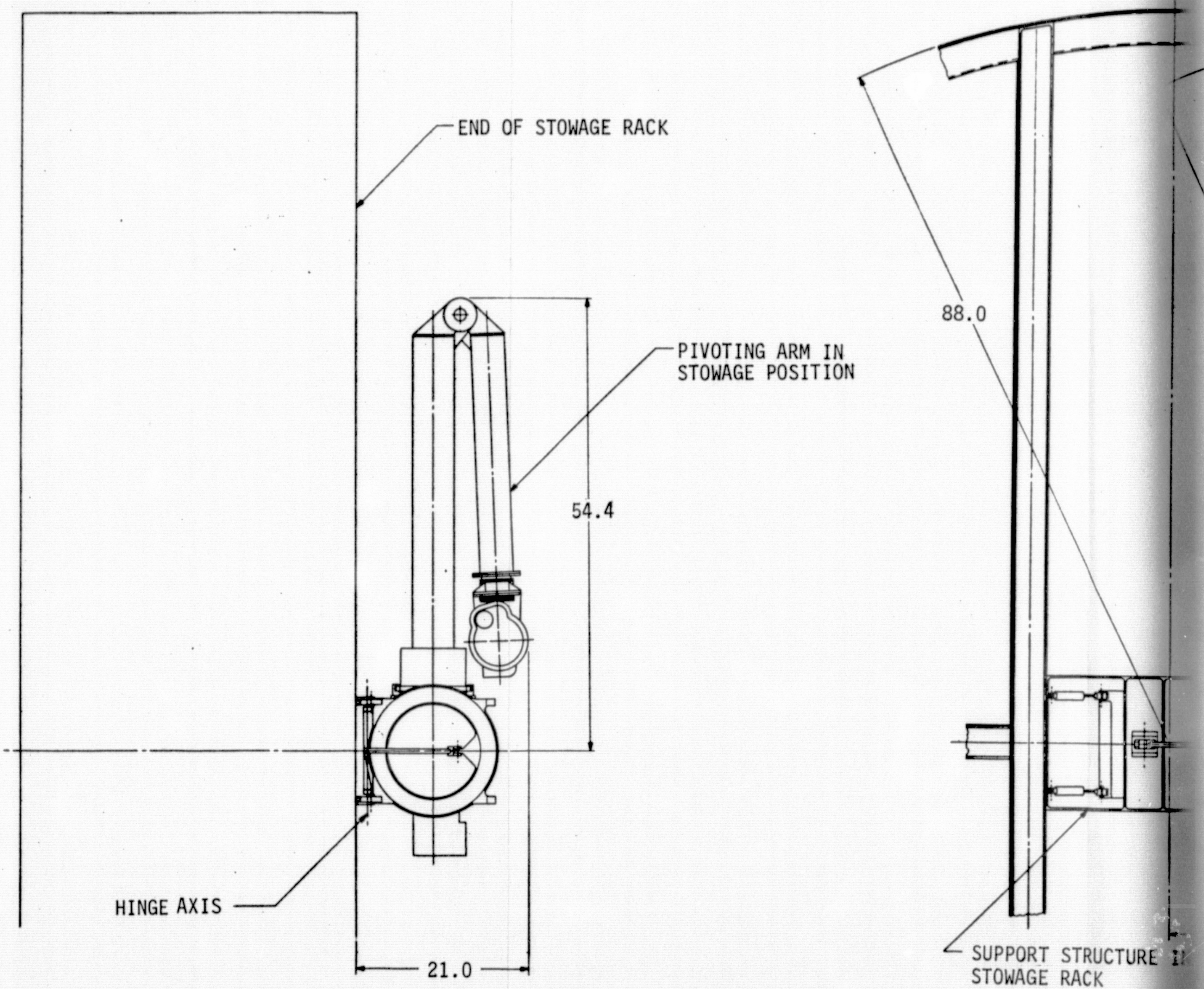
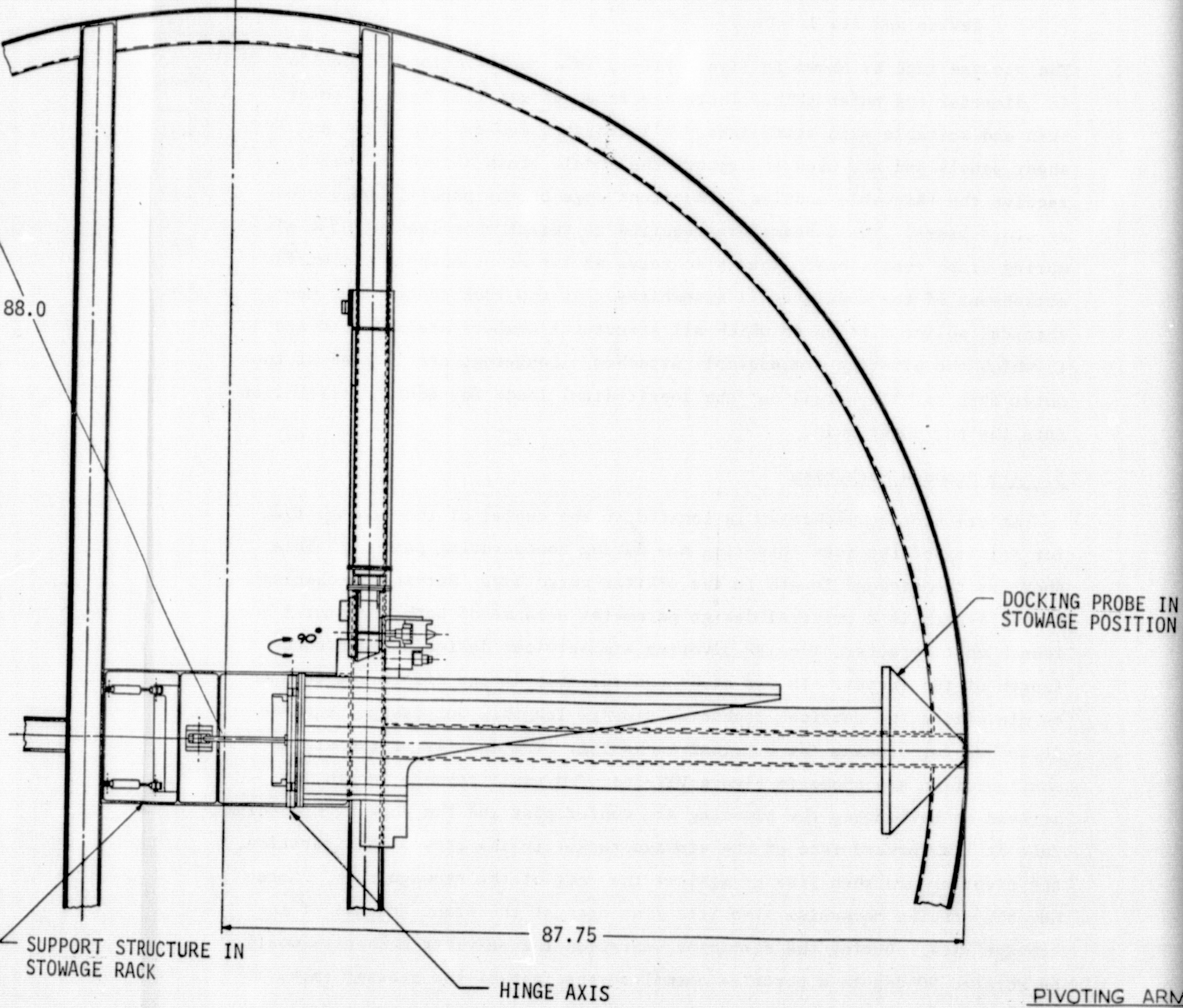


Figure VII-16 On-Orbit Servicer Stowed Configuration



PIVOTING ARM -
STOWAGE POSITIO

SCALE

FOLDOUT FRAME 2

itself. The design of the elbow joint permits this. Also the shoulder roll is driven to where the folded arm lies flat against the face of the stowage rack.

In the stowed configuration, the servicer mechanism is fastened to the stowage rack by latches at the center post, docking probe, arm elbow, and at end effector. These latches are to take launch, reentry and landing loads and vibrations. A single latch is used to hold the pivoting arm mechanism in the operating position. This latch along with its operating cylinder is shown on Figure VII-14. Gaseous nitrogen is stored in a cylinder to provide the pressure to operate the various latches. Sufficient nitrogen for several operating cycles can be provided. The pneumatic cylinders provide high forces for low weight. The mechanism for folding the arm is also nitrogen operated. The nitrogen storage tank can be recharged before each flight.

VIII. STS IMPACT ANALYSIS

As a necessary part of the study of the feasibility of the various maintenance concepts, the impacts on the elements of the STS were evaluated. The elements of the STS included primarily the orbiter and full-capability tug, although ground support impacts were also investigated. The main result of the STS impact analysis was that there were no major impacts identified. Several minor impacts were identified and are discussed below.

A. GROUND OPERATIONS

Most of the work performed during this study involved the investigation of performing the maintenance of spacecraft while in orbit. However, it should be realized that the expenses of maintenance occur on the ground, during development, production and operations. Figure VIII-1 presents a schematic of the full recurring cycle of potential maintenance missions and represents the entire process that was investigated and costed by us during this study. The figure illustrates some 32 different activities, only eleven of which (marked with asterisks) directly involve orbital maintenance. This illustrates how orbital maintenance is closely interwoven with STS operations.

In-depth assessments of ground and flight operational requirements as applicable to the STS elements and servicing hardware end items (spacecraft, servicers, replacement modules, etc) were summarized. These assessments were made for the purpose of identifying those operational and support requirements which are common or unique to specific maintenance modes and/or servicing concepts. Further evaluations of each requirement provided indications of those which are included in the basic costing work breakdown structure (WBS); the remaining requirements were investigated as imposing potential impacts on current STS designs or servicing program costs.

Each of the three maintenance modes--expendable, ground refurbishable, and on-orbit maintainable--were considered. The on-orbit maintainable mode was represented by the pivoting arm on-orbit servicer, EVA, and SRMS maintenance concepts. For each of these five maintenance concepts, those associated STS elements and servicing hardware end items were compared

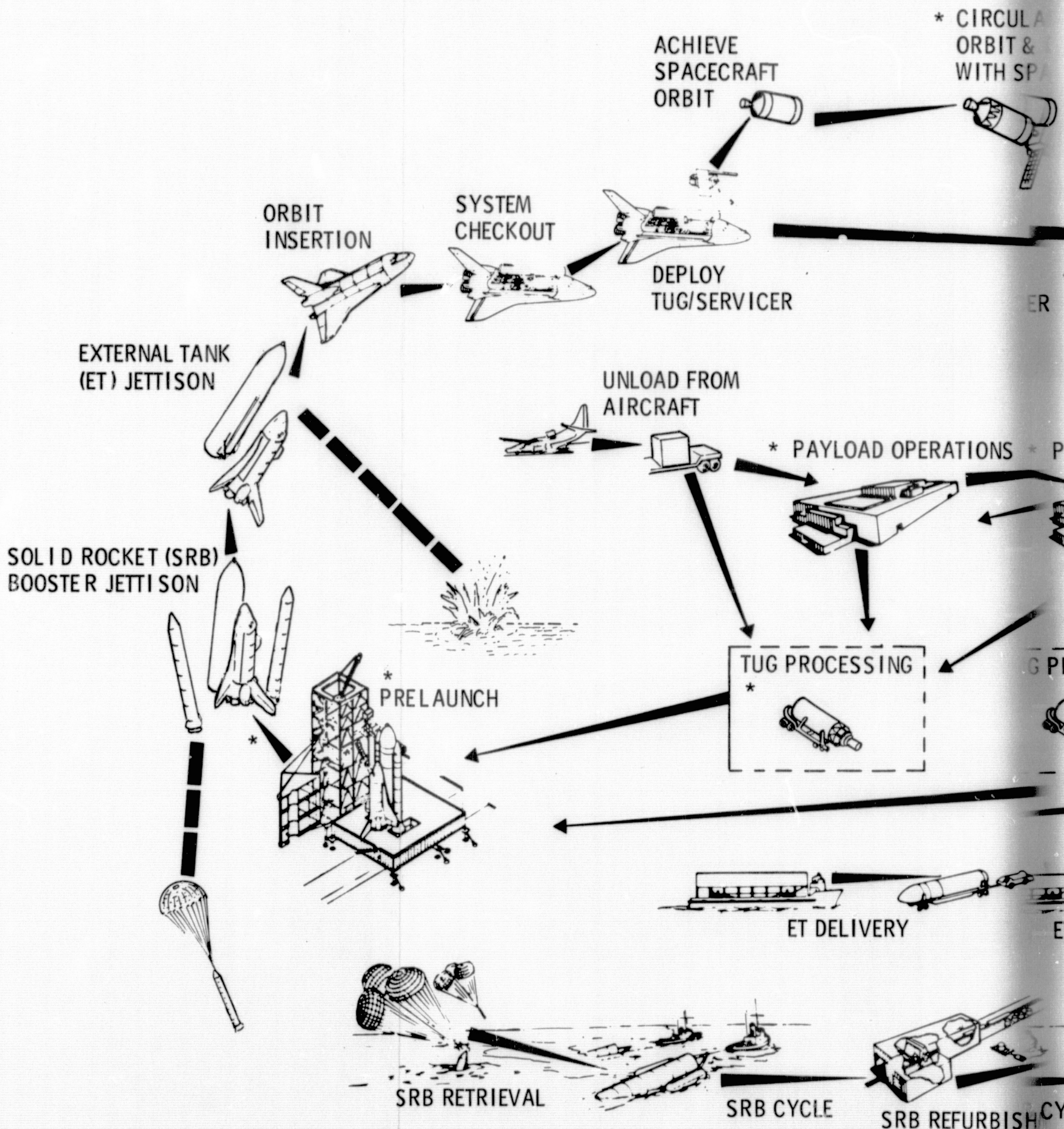


Figure VIII-1 STS Maintenance Recurring Cycle

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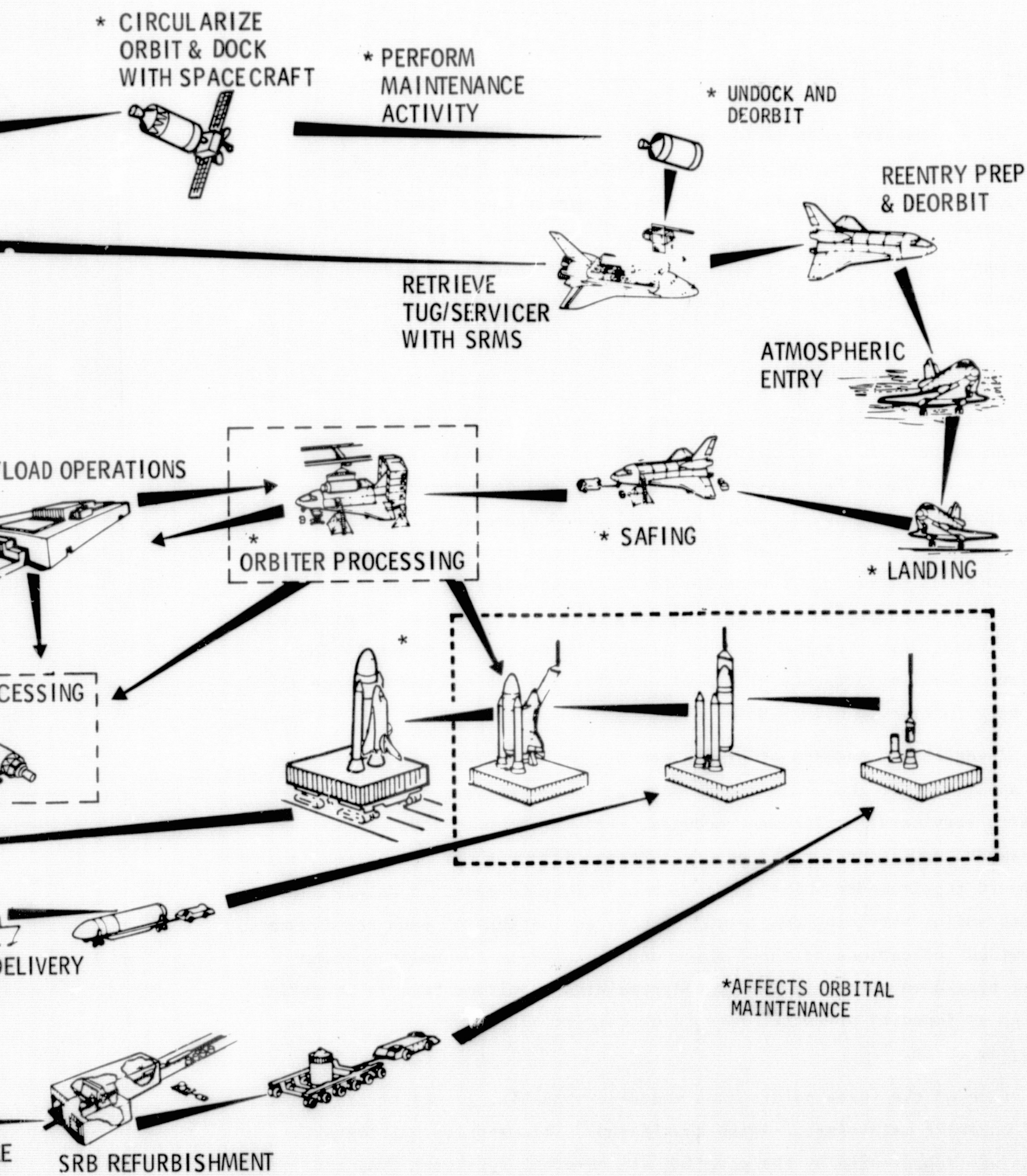

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with common listings of operational (ground, flight, and recovery) functions and support (ground equipment, logistics, and facilities) items. Resulting requirements were then compared for commonalities or uniqueness among the maintenance modes and servicing concepts. Those requirements found to be included in the work breakdown structure were considered as presenting no design or costing impacts; those not in the WBS were evaluated further to determine impact significance and/or magnitudes.

Results of the assessments indicated no significant impacts of ground and flight operational/support requirements on the STS elements or servicing hardware end items. This result is commensurate with the NASA intention to design the STS to be compatible with all three maintenance modes. However, requirements for additional bonded storage space at the launch site were identified, and the need for additional flight support equipment (support structures, special-purpose manipulator end effectors, and EVA tools) was noted. Also, the assessments revealed other considerations (spacecraft contamination, crew safety, docking and latching mechanism selection, and multiple spacecraft dockings) which must be taken into account during subsequent studies.

Each requirement identified during the STS impact analysis was defined in detail (quantities and weights of additional support items, additional training, additional personnel support, etc) and applied to subsequent cost support efforts (Chapter IX). Additional costs resulting from the requirements were reflected in the total comparative costs generated for each maintenance mode and servicing concept.

Ground operational steps included in the STS impact analysis are shown in Fig. VIII-2 which represents the expendable spacecraft case. The detailed operational flows and functions shown represent those activities and operational steps required at the respective launch sites (ETR and WTR) during ground processing of the:

(1) Orbiter,
(2) Tug,
(3) External tank,
(4) Solid rocket booster, and
(5) Spacecraft.

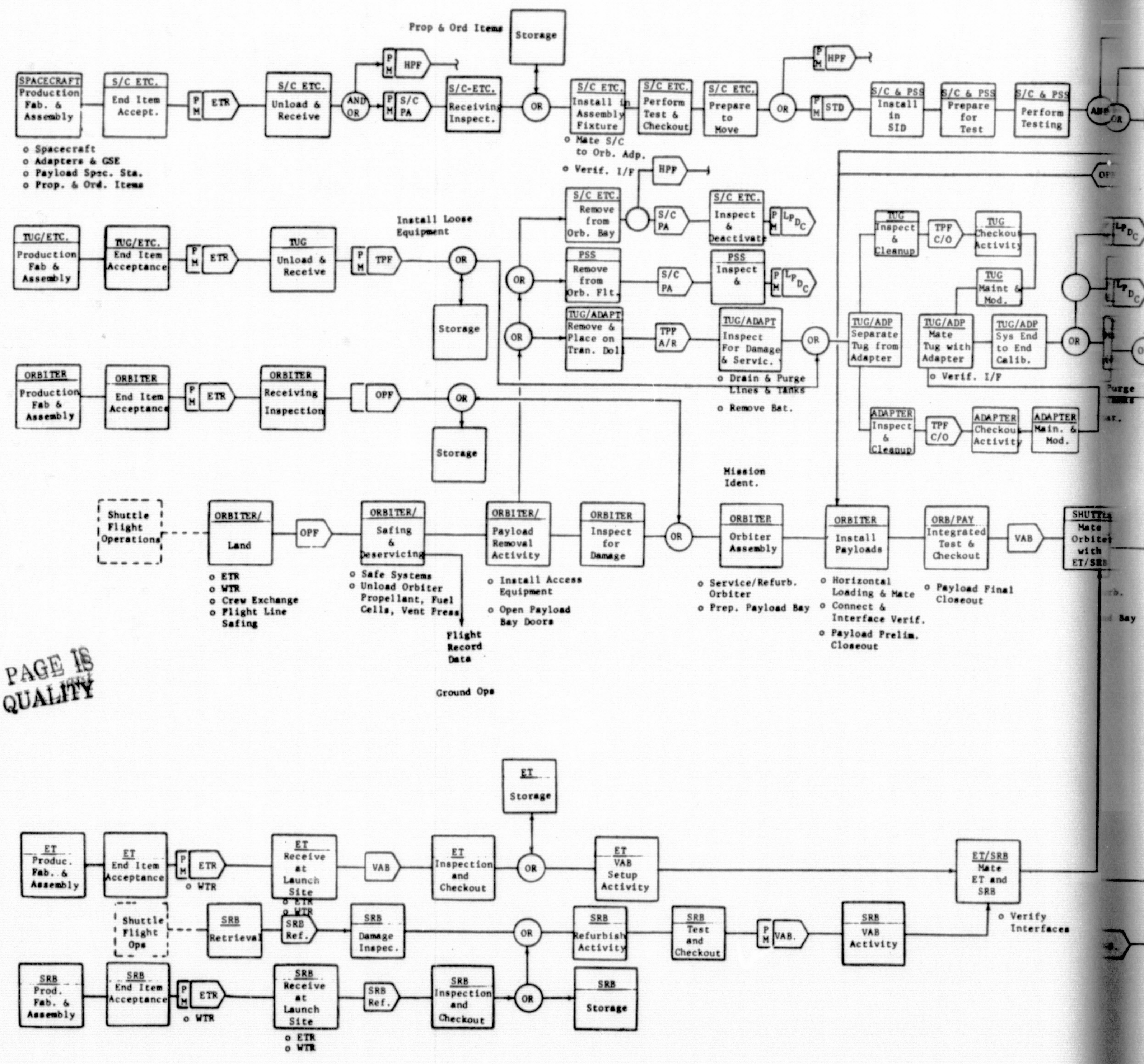

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Figure VIII-2 Ground Processing Flowchart (Baseline)

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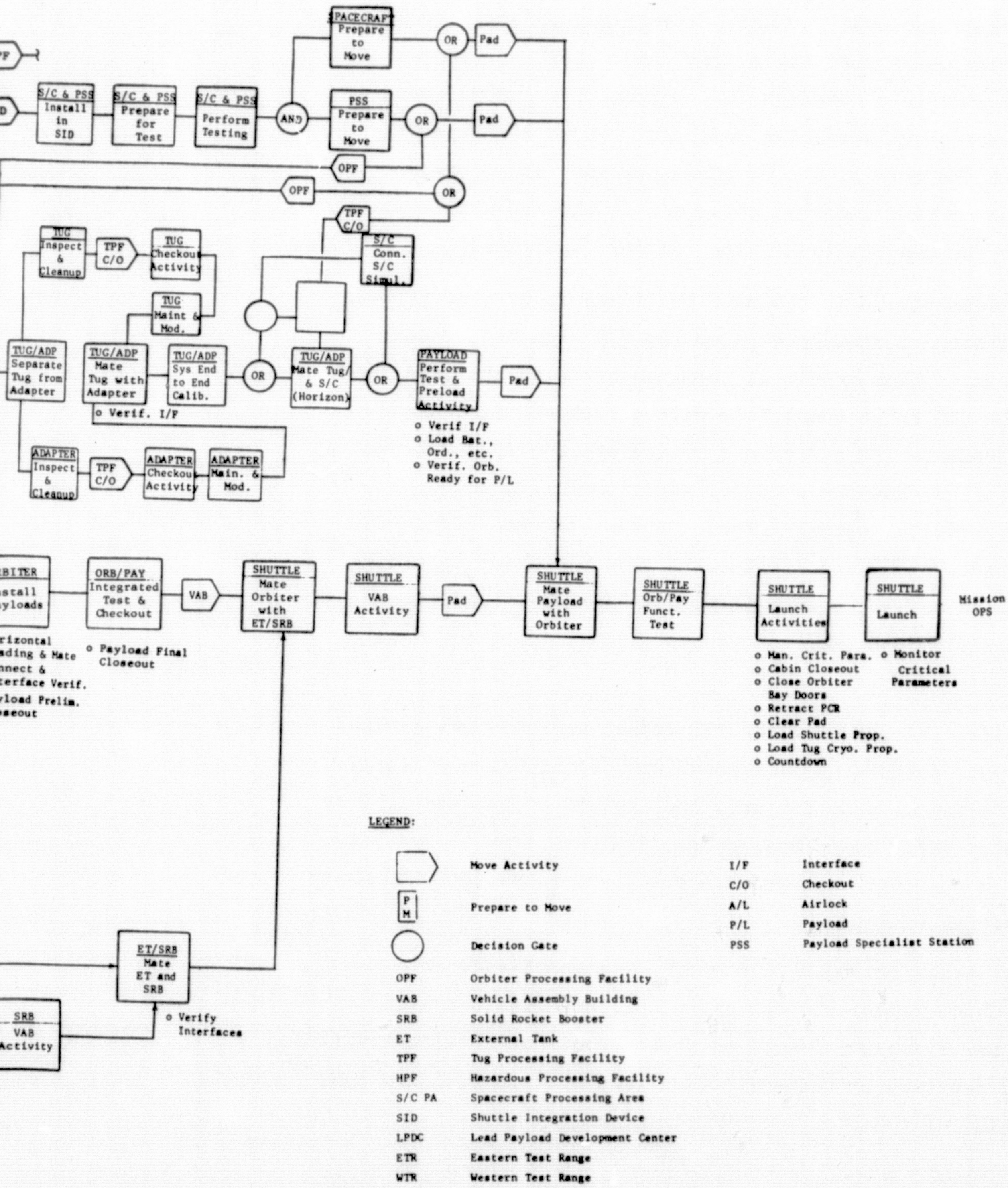

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Detailed assessments were made of each identified operational step in order to determine functional and support requirements and to establish requirement commonalities among the various hardware elements identified above. As the STS baseline flow is for the expendable spacecraft concept, it was only necessary to prepare a new assessment for the other four maintenance concepts. The operational elements considered are shown in Table VIII-1 grouped as to which are common and unique to the STS units.

Table VIII-1 Operational Elements Related to STS Units

	UNIQUE	COMMON
GROUND OPERATIONS	PRODUCTION, FABRICATION, ASSEMBLY, AND ACCEPTANCE LAUNCH OPERATIONS RECOVERY PROCEDURES REFURBISHMENT ACTIVITIES	RECEIVING INSPECTIONS FUNCTIONAL TESTING HARDWARE INSTALLATION INTEGRATED SYSTEMS TESTING
FLIGHT OPERATIONS	DEPLOY TUG, EOTS, OR FREE-FLYING SERVICER RETRIEVE FAILED SPACECRAFT FOR RETURN STOW SPACECRAFT FOR RETURN	RENDEZVOUS/DOCKING ON-ORBIT SERVICING
GROUND SUPPORT EQUIPMENT	ADAPTERS ORDNANCE TEST EQUIPMENT GUIDANCE AND NAVIGATION TEST EQUIPMENT	PORTABLE TRANSPORTERS MONITORING/TEST EQUIPMENT LATCH MECHANISM TEST EQUIPMENT ALIGNMENT EQUIPMENT
LOGISTICS	CRITICAL SPARES PAYLOAD REMOVAL INSTRUCTIONS	TRANSPORTATION/HANDLING INSTRUCTIONS INTERFACE CONTROL DOCUMENTS TRAINING/SIMULATION PROCEDURES FLIGHT PLANS/OPERATIONS CHECKLISTS
FACILITIES	HAZARDOUS PROCESSING FACILITIES COMMODITIES RECOVERY FACILITIES	PROCESSING/TEST FACILITIES CLEAN ROOM FACILITIES BONDED STORAGE AREAS

As assessment efforts progressed, additional ground flows developed for the on-orbit servicer, and replacement modules were incorporated into the baselined flowchart. An illustration of the incorporated ground flow is presented in Figure VIII-3 for the on-orbit servicer. This simplified figure, as compared to Figure VIII-2, shows the major interaction points between on-orbit servicers and the baseline ground processing flow. It may be noted that the flow corresponds closely to the previously baselined flow for an expendable spacecraft, and its integration into the flowchart presented no significant impacts on either STS elements or program costs.

Other maintenance concepts (such as servicing by EVA or by use of the orbiter SRMS) were similarly evaluated, and their ground processing flows

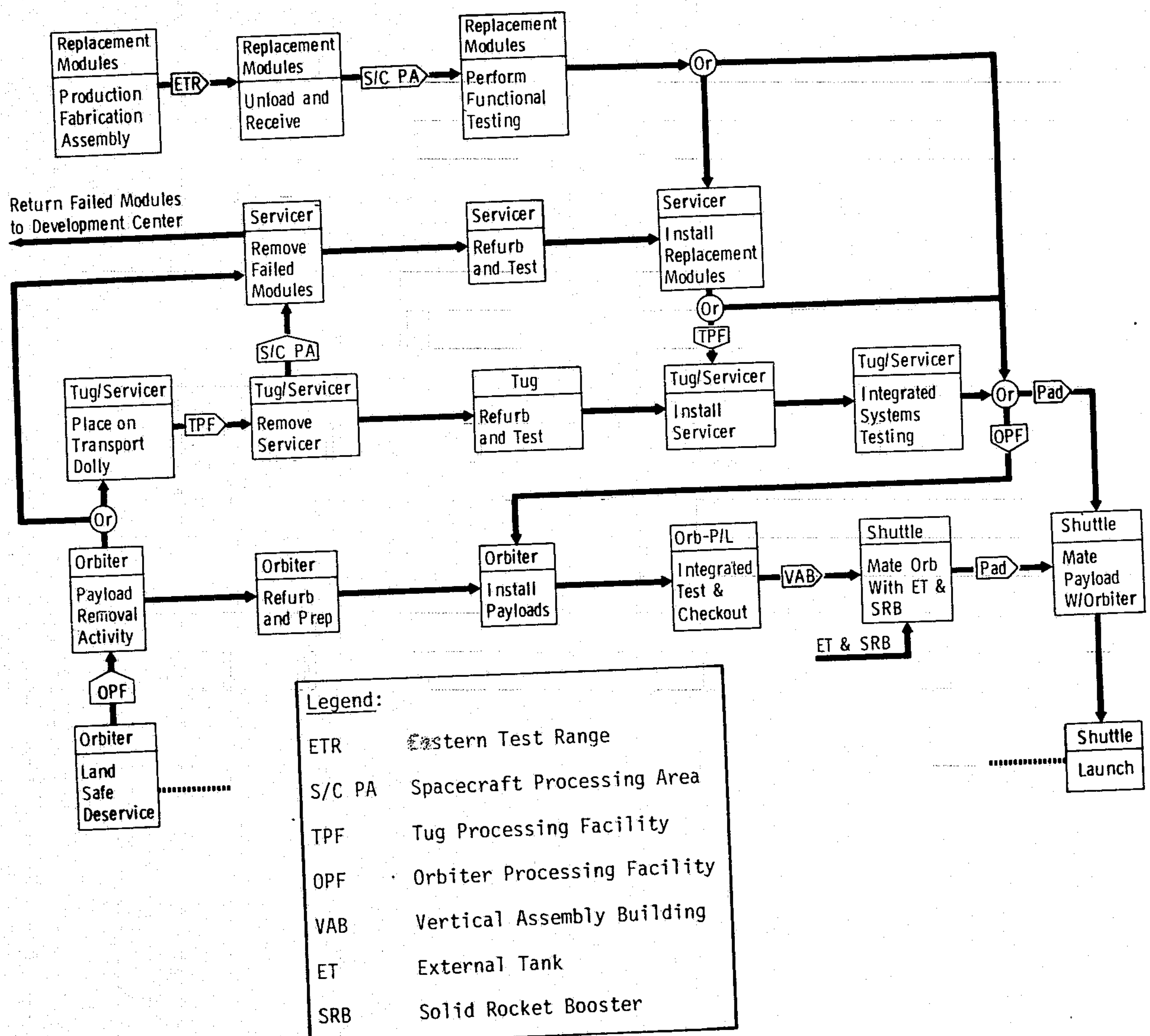

Figure VIII-3 On-Orbit Servicer Ground Processing Flow

were found to follow the same general flows as developed for the on-orbit servicer. The flows for the EVA and SRMS servicing concepts were found to be somewhat less complex than for the on-orbit servicer in that there are no operational interfaces with the tug. The flow for the on-orbit servicer could also be shown without the tug for missions where the on-orbit servicer is used to maintain LEO spacecraft.

During the impact analysis, considerations were also given to those major flight operational functions which could potentially impact the STS elements or servicing hardware end items. Each major function was assessed in relation to each element and end item to determine commonalities among the elements/end items or to identify unique operations which could present potential impacts on STS element or servicer designs.

B. STS IMPACTS

Although no major impacts were identified, Tables VIII-2, VIII-3, and VIII-4 present a summary of the orbiter, tug, and ground system impacts identified. These impacts are above the baseline capabilities of the STS as identified in Chapter III. They were identified from the totality of the contract work and not just those discussed in this chapter. The functional and hardware requirements of Chapter III provided a useful crosscheck.

C. CARGO BAY UTILIZATION

Stowage of typical payload complements in the orbiter cargo bay was examined to identify any problems that might exist. Typical payload complements of on-orbit servicers, characteristic set spacecraft, orbital maneuvering system (OMS) kits and tugs were considered for the LEO and HEO cases. This check was for the stowed configurations of the spacecraft as opposed to the servicing operations considerations of Chapter IV. It was concluded that there is adequate room in the orbiter cargo bay for the small payload combinations considered and that spacecraft in the maximum STS efficiency configuration further decreased cargo bay stowage problems.

The LEO case is shown in Figure VIII-4 which includes the OMS kit, the SPAR/DSMA cargo bay only servicer and alternate stowage of the BESS, two GRAVSATS, or the LXRT. The lower half of the figure is for the SSPD expendable configuration of spacecraft while the upper half is for

Table VIII-2 Orbiter Impacts Summary

GROUND REFURBISHMENT	PIVOTING ARM	EVA	SRMS
SPACECRAFT EXCHANGE CAPABILITY CONTAMINATION CONTROL	CREW CONTROL & DISPLAY STATION STOWAGE RACK IN BAY THERMAL CONTROL OF MODULES CAUTION AND WARNING POWER DATA, FLUIDS, & COMMAND INTERFACES PAYLOAD BAY VIDEO SYSTEM	SPACECRAFT SUPPORT PLATFORM CREW SAFETY REQUIREMENTS STOWAGE RACK IN BAY SPECIAL PURPOSE EVA TOOLS & END EFFECTORS ADDITIONAL TRANSLATION AIDS & TETHERS LIGHTING CONTAMINATION CONSIDERATIONS THERMAL CONTROL OF MODULES ADDITIONAL EVAs & EVA CONSUMABLES POWER DATA, FLUIDS, & COMMAND INTERFACES	CREW CONTROL & DISPLAY STATION ADDITIONAL CAPABILITY TO BASELINE SRMS SPACECRAFT SUPPORT PLATFORM STOWAGE RACK IN BAY SPECIAL PURPOSE END EFFECTOR POSSIBLE ADDITIONAL ARM THERMAL CONTROL OF MODULES POWER DATA, FLUID, & COMMAND INTERFACES

Table VIII-3 Tug Impacts Summary

GROUND REFURBISHMENT	PIVOTING ARM	EVA	SRMS
SPACECRAFT EXCHANGE CAPABILITY REQUIRES 2 TUGS FOR SOME SPACECRAFT IN GEOSTATIONARY	DOCKING SYSTEM INTERFACE WITH SERVICER MECHANISM SPACECRAFT INTERFACES FROM TUG THROUGH SERVICER, TO SPACECRAFT IF PIVOTING ARM CARRIED ROUND TRIP, OTHER SPACECRAFT CAPABILITIES REDUCED AS FOLLOWS: FOR GEOSTATIONARY WAS IS DEPLOY 7926 7000 RETRIEVE 3396 3200 ROUND TRIP 2500 1920 MODULES MUST ALSO BE CARRIED UP CAPABILITY TO DUMP MODULES BEFORE LEAVING GEOSTATIONARY POWER DATA, FLUID, & COMMAND INTERFACES VIDEO SYSTEM THERMAL CONTROL OF SERVICER, MODULES	CREW SAFETY IF EVA PERFORMED WITH TUG IN BAY LEO SPACECRAFT SUPPORT PLATFORM AND STOWAGE RACK VOLUME CONSTRAINTS	VOLUME & REACH RESTRAINTS IF SRMS USED FOR MAINTENANCE WITH TUG IN BAY LEO SPACECRAFT SUPPORT PLATFORM & STOWAGE RACK VOLUME CONSTRAINTS

Table VIII-4 Ground and Other Impacts

GROUND REFURBISHMENT	PIVOTING ARM	EVA	SRMS
BONDED STORAGE AREA FOR SPACECRAFT	BONDED STORAGE AREA FOR MODULES ADDITIONAL GSE SPECIAL PURPOSE SIMULATORS GROUND INVENTORY LOGISTIC PROGRAM GROUND CONTROL & DISPLAY AREA FOR SERVICER OPERATION SERVICER TRAINERS & MOCK-UPS	BONDED STORAGE AREA FOR MODULES ADDITIONAL NEUTRAL BUOYANCY TRAINING ADDITIONAL GSE GROUND INVENTORY LOGISTICS PROGRAM	BONDED STORAGE AREA FOR MODULES ADDITIONAL GSE GROUND INVENTORY LOGISTICS PROGRAM

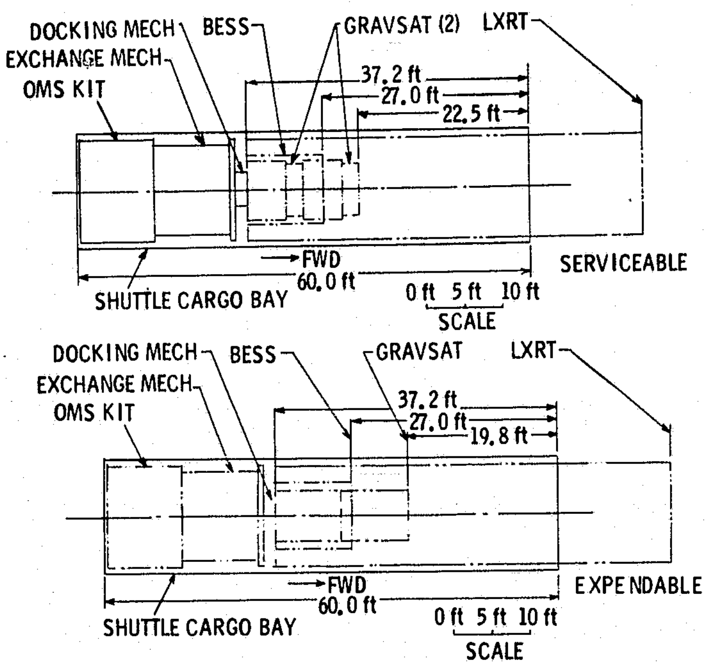

Figure VIII-4 Cargo Bay Utilization - LEO

space-servicable maximum STS efficiency spacecraft configurations. The small spacecraft fit very well with plenty of room, but the LXRT will not fit when the OMS or cargo bay only exchanger mechanism is carried. The servicable form of GRAVSAT saved some additional length, but this did not occur for the BESS or LXRT.

The comparable case for HEO is shown in Fig. VIII-5 which includes a full capability tug, the pivoting arm on-orbit servicer, and alternate stowage of the upper atmosphere explorer (UAE), international communications satellite (INTELSAT), and environmental monitoring satellite (EMS). Any of these characteristic set spacecraft can be combined and flown with the pivoting arm servicer from a volume point of view. In this case there are significant cargo bay length savings associated with the maximum STS efficiency serviceable spacecraft configurations. Note also that the pivoting arm on-orbit servicer uses up less of the cargo bay length than any of the expendable configuration characteristic set spacecraft.

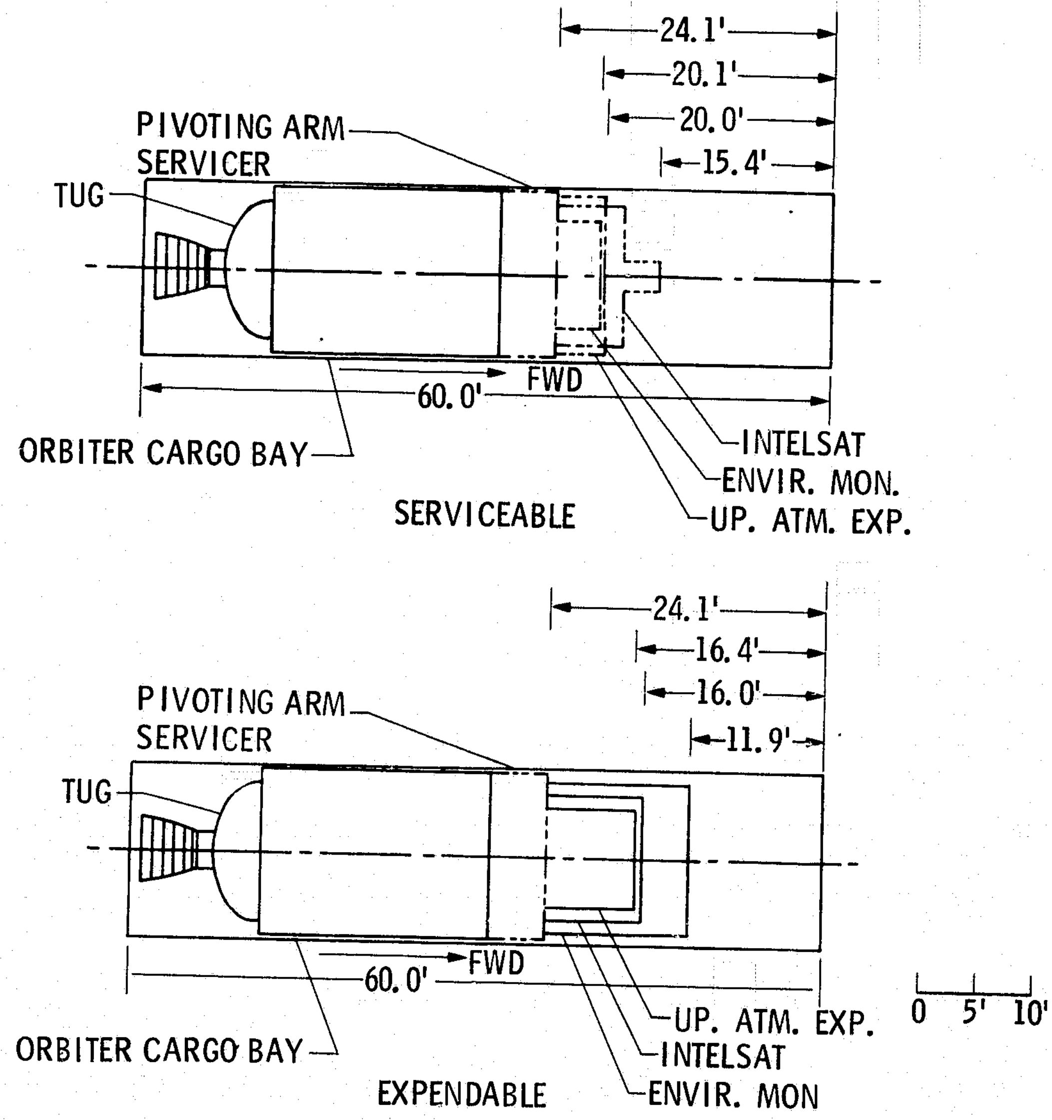

Figure VIII-5 Cargo Bay Utilization - HEO

D. OTHER CONSIDERATIONS

During the performance of the STS impact analysis, a few areas were identified where special considerations will subsequently be required. Although the areas of concern do not present impacts on the current STS element designs, they may influence future efforts associated with the development of operational or procedural constraints for each servicing mission. The areas of concern identified during the analysis were:

1. Contamination - operational or procedural constraints on orbiter venting or thruster firings may be necessary to preclude the

contamination of critical spacecraft surfaces and hardware during servicing operations while the spacecraft is in or adjacent to the orbiter.

2. Spacecraft Configuration - design considerations may be necessary relative to the retraction (and subsequent redeployment) of spacecraft appendages (antennas, protuberances, etc) to permit placement of the spacecraft in the orbiter cargo bay (or alongside the orbiter) during servicing operations.

3. Spacecraft Design for EVA - design considerations will be necessary to assure that serviceable spacecraft meet design requirements for EVA servicing. These considerations will also apply to the incorporation of required crew tanslation aids and restraints.

4. Docking Mechanism - consideration will be necessary to assure that docking mechanisms on serviceable spacecraft and the various servicers are universally compatible. This consideration should be extended to cover compatibilities between the spacecraft replacement modules and their associated module exchange mechanism(s).

   The configuration of the docking mechanism can strongly affect the configurations of the spacecraft and of the on-orbit servicer. This point was addressed with respect to the servicer at some length in Chapter IV, where adaptability of the servicer to either central or peripheral docking mechanisms was used as a design criteria. The selected pivoting arm servicer can be adapted to central or peripheral docking systems. However, there are some limitations. These are illustrated by Fig. III-6 which shows a square frame docking mechanism and its relationship to a typical spacecraft and a pivoting arm servicer as represented by a stowage rack location. The interference between the docking mechanism legs and crossbars and the end of the spacecraft can be easily seen. About one-half the useable axial module replacement volume is lost for large diameter spacecraft and somewhat more for small diameter spacecraft.

5. Servicing Monitors - consideration will be necessary to assure that adequate monitoring devices (data displays, video monitors, etc) are incorporated to assist the crewmen and ground support personnel in the monitoring of servicing operations.

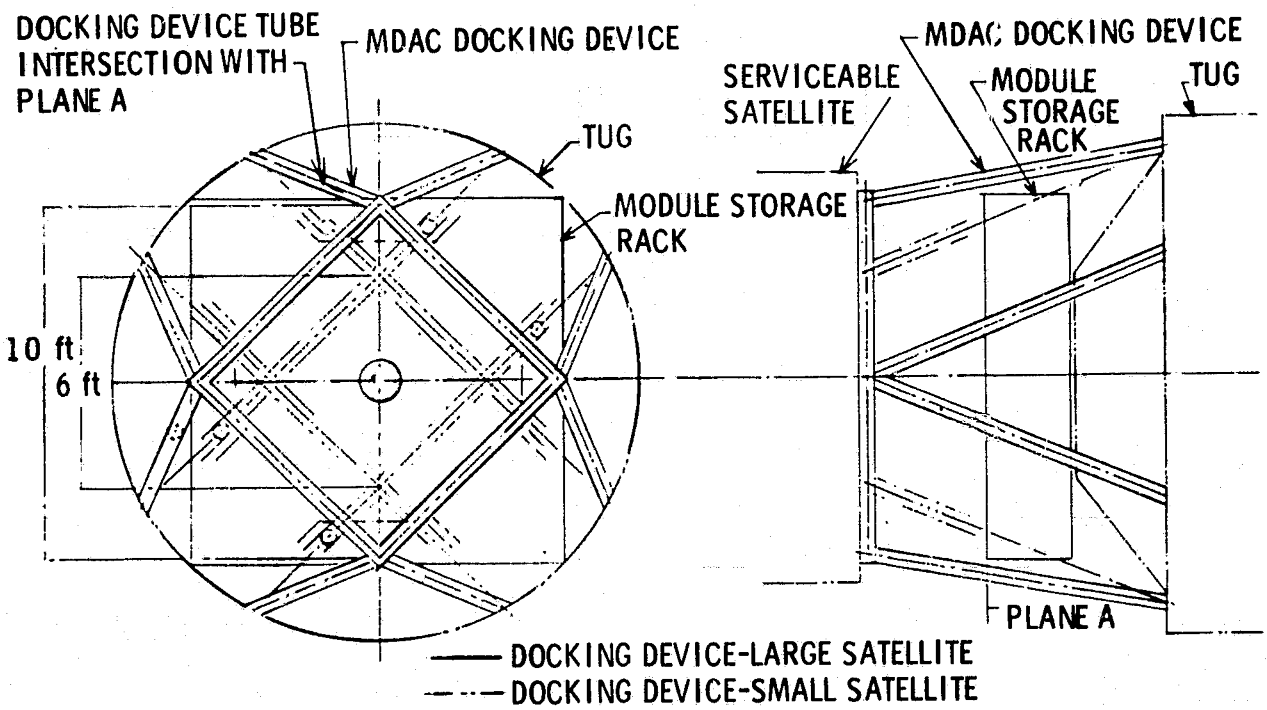

Figure VIII-6 Square Frame Docking Mechanism Interface

6. <u>Contingency Planning</u> - extensive planning and procedural development activities will be necessary to cover any contingencies encountered during the various servicing modes and concepts.

It should be re-emphasized that the foregoing considerations do not impact the current STS element design efforts; however, they are believed to be of significance when future serviceable spacecraft programs are developed.

IX. COST GENERATION AND ANALYSIS

One of the most important facets of this study involved the determination of the economic benefits to NASA and to the user community of developing the capability to perform maintenance on spacecraft of the shuttle era. This involved not only the determination of the economic benefits of maintenance, in general, but also the determination of which mode, concept, and system will provide the greatest economic benefits, and for which spacecraft programs. In order to accomplish these evaluations, costs were developed for flying the shuttle era automated spacecraft program in the three competing modes of expendable, ground refurbishable, and on-orbit maintainable. Costs were also developed for performing on-orbit maintenance using EVA, SRMS, and an on-orbit servicer mechanism. Costs were developed to compare the two best servicer mechanisms, a pivoting arm-axial module removal-servicer, and a general purpose manipulator-radial module removal-servicer.

Results of the economic analyses showed that the greatest economic benefits to NASA and to the user community would come from the earliest possible development and use of a pivoting arm servicer mechanism. Over 9 billion dollars can be saved during the shuttle era with an early development and early user acceptance of a pivoting arm servicer.

This chapter will discuss the approach, assumptions, costing techniques and equations, development of data, and results used to obtain this set of conclusions.

A. APPROACH

An important part of the understanding of the techniques and results of the economic analysis is a knowledge of the ground rules used. Table IX-1 presents a brief summary of some of the more important ground rules. The first rule presented - a constant availability across all three maintenance modes - was used to establish the basis of the costing analysis and is discussed in more detail below.

Table IX-1 Cost Estimation Ground Rules

• AVAILABILITY AND SPACECRAFT PROGRAM DURATION HELD CONSTANT ACROSS ALL THREE MAINTENANCE MODES
• ALL COSTS ARE 1975 $
• STS COST PER FLIGHT (ORBITER - $12.0 MILLION, FULL CAPABILITY TUG - 1.1 MILLION)
• SHUTTLE IOC 1980 AT ETR
• SHUTTLE OPERATIONAL FROM WTR - 1983
• FULL CAPABILITY TUG IOC-DECEMBER 1983
• LEO SPACECRAFT LAUNCHED ON EXPENDABLE LAUNCH VEHICLES IN 1979 CONSIDERED FOR SERVICING WITH ORBITER STARTING IN 1980
• MEO AND HEO SPACECRAFT LAUNCHED WITH INTERIM TUGS IN 1982 AND 1983 CONSIDERED FOR SERVICING WITH ORBITER/FULL CAPABILITY TUG STARTING IN 1984
• SPACECRAFT PROGRAM COSTS CALCULATED FOR ALL THREE MODES FROM 1979-1991 FOR LEO SPACECRAFT AND FROM 1982-1991 FOR MEO AND HEO SPACECRAFT
• MAINTENANCE PERFORMED ON AUTOMATED SPACECRAFT FLOWN DURING SHUTTLE ERA
• NO DOD OR SORTIE LAB SPACECRAFT CONSIDERED FOR MAINTENANCE

Figure IX-1 presents a brief flow schematic depicting the performance of the cost analysis. In performing the total economic analyses, a "quick-look", preliminary analysis was performed to determine the main cost drivers, and then a detailed "baseline" analysis was performed using the best data to obtain the most accurate results. This was followed by a sensitivity study to determine the accuracy and validity of input data, the effects of data inputs on results, and the effects of possible future changes in input data on the study results.

The general approach used to perform the costing involved the establishment of a Work Breakdown Structure (WBS) which was used to help cost the three modes. The WBS was established so that all of the maintenance modes, maintenance concepts, and spacecraft programs could be costed using the same WBS. Figure IX-2 presents the WBS used and Table IX-2 presents the WBS dictionary explaining the items in the WBS.

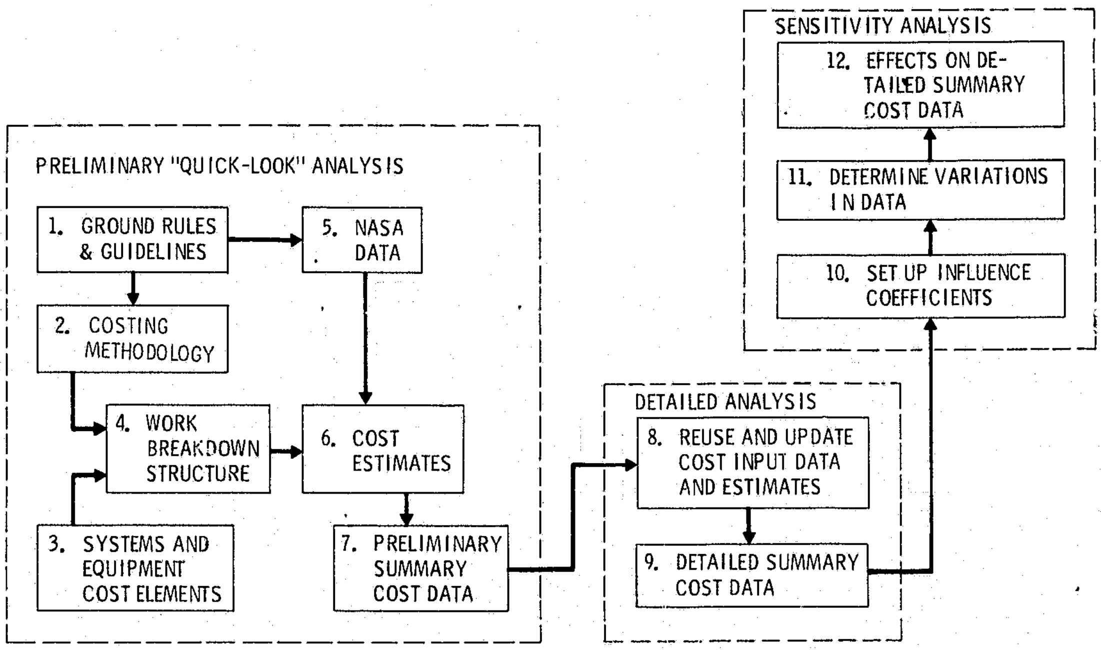

Figure IX-1 Cost Analysis Performance

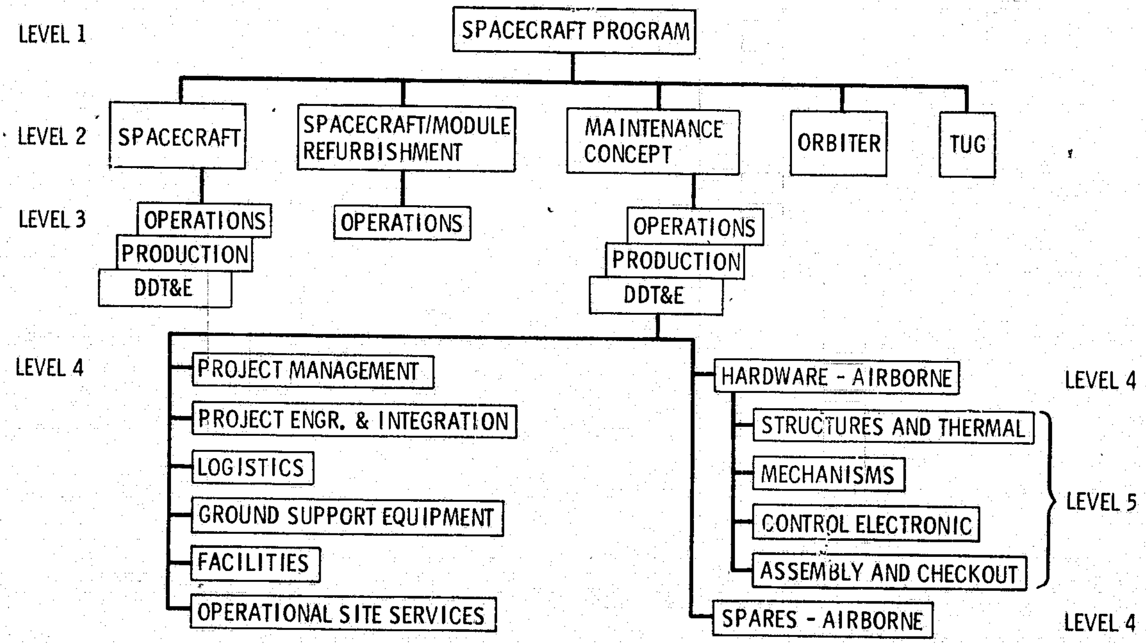

Figure IX-2 Work Breakdown Structure

Table IX-2 Work Breakdown Structure Dictionary

WBS TITLE: SPACECRAFT PROGRAM — WBS LEVEL 1

This element summarizes those projects (spacecraft, spacecraft/module refurbishment, maintenance concept, orbiter and tug), as applicable, to accomplish expendable, ground refurbishable and on-orbit maintainable spacecraft programs.

WBS TITLE: SPACECRAFT — WBS LEVEL 2

WBS Number: 10

This element summarizes the DDT&E, production and operations phases. Each phase includes the direct and indirect effort to provide hardware, software, services and facilities as required.

WBS TITLE: SPACECRAFT-DDT&E — WBS LEVEL 3

WBS Number: 11

This element consists of the one time cost of design, development, testing and evaluation of spacecraft hardware. Specifically, it includes the following: development engineering and development support, major test hardware, captive and ground test, ground support equipment, tooling and special test equipment, site activation and other program peculiar costs not associated with repetitive production.

WBS TITLE: SPACECRAFT-PRODUCTION — WBS LEVEL 3

WBS Number: 12

This element consists of the costs associated with producing flight hardware through acceptance of the hardware by the government including all costs to: fabricate, assemble, checkout and acceptance test of flight hardware, spares, and maintenance of tooling and special test equipment.

WBS TITLE: SPACECRAFT-OPERATIONS — WBS LEVEL 3

WBS Number: 13

This element consists of the repetitive services and activities that comprise launch

WBS Number: 13 (Continued)

operations. Launch operations include spacecraft receiving and inspection, prelaunch checkout, orbiter/tug mating and checkout, and launch countdown. Flight operations include mission planning, flight control, data reduction, analysis and documentation.

WBS TITLE: SPACECRAFT/MODULE REFURBISHMENT — WBS LEVEL 2

WBS Number: 20

This element consists of restoring failed spacecraft/modules, retrieved from orbit, to a flight readiness condition for subsequent missions. This effort includes removal from the orbiter, shipment to the vendor's facility, inspection, disassembly, maintenance, refurbishment and acceptance testing, and shipment to launch site.

WBS TITLE: SPACECRAFT/MODULE REFURBISHMENT-OPERATIONS — WBS LEVEL 3

WBS Number: 23

Same as Spacecraft/Module Refurbishment

WBS TITLE: MAINTENANCE CONCEPT — WBS LEVEL 2

WBS Number: 30

Same as Spacecraft

WBS TITLE: MAINTENANCE CONCEPT-DDT&E — WBS LEVEL 3

WBS Number: 31

Same as Spacecraft-DDT&E

WBS TITLE: MAINTENANCE CONCEPT-PRODUCTION — WBS LEVEL 3

WBS Number: 32

Same as Spacecraft-Production

WBS TITLE: MAINTENANCE CONCEPT-OPERATIONS — WBS LEVEL 3

WBS Number: 33

This element summarizes the cost of launch operations, flight operations and refurbishment including all management and supporting functions.

WBS TITLE: PROJECT MANAGEMENT — WBS LEVEL 4

WBS Number: 3X1

This element summarizes the activities of cost/performance management, configuration management, information management and GFE management required to accomplish overall project objectives.

WBS TITLE: PROJECT ENGINEERING AND INTEGRATION — WBS LEVEL 4

WBS Number: 3X2

This element summarizes the activities of analysis and integration, shuttle interface, payload interface, reliability, quality assurance, safety and human engineering required to direct and control the design.

WBS TITLE: HARDWARE-AIRBORNE — WBS LEVEL 4

WBS Number: 3X3

This element summarizes the subsystems of a vehicle system and its assembly and checkout. Each subsystem includes the design, development test, qualification test of components and subsystems, tooling, procurement, hardware fabrication, quality control, assembly and checkout efforts which satisfy applicable design requirements.

WBS TITLE: STRUCTURES AND THERMAL — WBS LEVEL 5

WBS Number: 3X31

This element is the load carrying entity which provides mounting and supporting surfaces for all equipment. It consists of the body or primary structure, secondary structures such as brackets, mounts, fairings, pyrotechnics and the thermal protection and insulation systems installed on the structural components.

WBS TITLE: MECHANISMS — WBS LEVEL 5

WBS Number: 3X32

This element is the mechanical or electromechanical devices of the on-orbit servicing system that will repetitively move objects from one point to another and may involve latch/attach operations.

WBS TITLE: CONTROL ELECTRONICS — WBS LEVEL 5

WBS Number: 3X33

This element is the electrical and electronic signal conditioning equipment of the on-orbit servicing system except those contained within a spacecraft module or available as STS baseline equipments.

WBS TITLE: ASSEMBLY AND CHECKOUT — WBS LEVEL 5

WBS Number: 3X34

This element summarizes the activities and materials required to perform final assembly, checkout and acceptance testing of the completed system.

WBS TITLE: SPARES-AIRBORNE — WBS LEVEL 4

WBS Number: 3X4

This element summarizes the effort to fabricate and maintain subsystem spares necessary to support the servicing and maintenance of system hardware at test and launch sites.

WBS TITLE: LOGISTICS — WBS LEVEL 4

WBS Number: 3X5

This element summarizes the effort to develop, implement and maintain a logistics activity to support the system hardware and includes maintainability analysis, spares management, analysis of support requirements, inventory management, warehousing and storage, training requirements and equipment, technical manuals and transportation analyses and planning.

WBS TITLE: GROUND SUPPORT EQUIPMENT — WBS LEVEL 4

WBS Number: 3X6

This element summarizes the design, production, software and maintenance activities for equipment required during fabrication effort and at the launch site to checkout, test, service, handle, transport, maintain and refurbish the airborne hardware.

WBS TITLE: FACILITIES — WBS LEVEL 4

WBS Number: 3X7

This element covers facilities (new or modification to existing) for fabrication, test, launch, maintenance and refurbishment of an operational program. Use of basic launch and operations facilities planned for the orbiter (if appropriate) is not included. The facilities effort includes planning, acquisition or modification and maintenance.

WBS TITLE: OPERATIONAL SITE SERVICES — WBS LEVEL 4

WBS Number: 3X8

This element summarizes the repetitive services and activities of launch operations, flight operations and maintenance/refurbishment. Launch operations include vehicle receiving and inspection, prelaunch checkout, orbiter/tug mating and checkout and launch countdown. Flight operations include mission planning, flight control, data reduction, analysis and documentation. Maintenance/refurbishment is restoring the reusable airborne vehicle, after each mission, to a readiness condition for subsequent missions. All costs pertaining to the vehicle inspection, maintenance/refurbishment, both scheduled and unscheduled, testing and checkout are included.

The work breakdown structure served as a common framework for all cost estimates. Continuity and correlation of cost data across the various spacecraft and maintenance concepts was assured by means of this WBS.

Cost data for each program element, except the orbiter and tug, were developed to WBS levels as follows:

1) Spacecraft — Level 3
2) Spacecraft/module refurbishment — Level 3
3) Maintenance concept
   - Nonhardware — Level 4
   - Hardware — Level 5

Orbiter and tug costs are operations costs and a cost per launch for each was supplied by MSFC.

Following the establishment of the WBS, a generalized format was established to compare the costs of the three modes according to the WBS. Figure IX-3 presents the general organization of the cost estimation which was used throughout the study to establish cost comparisons.

Spacecraft Program		Expendable	Ground Refurbishable	On-Orbit Maintainable
Orbiter		Launch S/C	Launch, Retrieve & Relaunch S/C	Launch S/C, Launch & Return Servicer & Modules
Tug		Launch S/C	Launch, Retrieve & Relaunch S/C	Launch S/C, Launch & Return Servicer & Modules
Spacecraft	DDT&E	Basic	Modified Basic	Modified Basic
	Production	Basic	Modified Basic for Fleet Size	Modified Basic for Fleet Size
	Operations	Launch C/O, Sustaining	Launch C/O for Fleet Size	Launch C/O for Fleet Size
Spacecraft/Module Refurbishment	Operations	N/A	Refurbish S/C, Launch C/O of Refurbished S/C, Sustaining	Replace Modules, Launch C/O of Modules, Sustain-ing
Maintenance Concept	DDT&E	N/A	N/A	Develop for Servicer
	Production	N/A	N/A	Develop for Servicer
	Operations	N/A	N/A	Launch C/O Servicer, Sustaining Servicer

Figure IX-3 Cost Estimation - Organization

Costing equations for each of the modes were then established to match the WBS cost format. Figures IX-4, IX-5, and IX-6 present the costing equations used for the three modes of expendable, ground refurbishable, and on-orbit maintainable. They are presented in the respective WBS formats and show the various cost parameters which were input. Table IX-3 presents a definition of all the cost parameters used.

Table IX-3 Costing Parameters

PARAMETERS - SAME FOR ALL MODES	PARAMETERS - DIFFERENT FOR EACH MODE: EXPENDABLE	GROUND REFURBISHABLE	ON-ORBIT MAINTAINABLE
n - NUMBER OF OPERATING CYCLES (NUMBER OF EXPENDABLE SPACECRAFT)	C_{KPE} - SUSTAINING	LF_1 - LOSS FACTOR - RETRIEVE	
n_f - ON-ORBIT FLEET SIZE		LF_2 - LOSS FACTOR, S/C	LF_3 - LOSS FACTOR, S/C
$C_{S/C}$ - SPACECRAFT UNIT COST		RF - REPLACEMENT S/C	pf - PARTS FACTOR
C_{NR} - SPACECRAFT DDT&E		PF - PARTS FACTOR	sf - FACTOR TO MODIFY S/C UNIT COST
LF - LOSS FACTOR, EMPLACE		SF - FACTOR TO MODIFY S/C UNIT COST	d - FACTOR TO MODIFY DDT&E
R - RATIO OF LAUNCH C/O TO S/C COST		D - FACTOR TO MODIFY DDT&E	C_{KPO} - SUSTAINING
		C_{KPG} - SUSTAINING	C_{S1} - SERVICER PRODUCTION
			C_{S2} - SERVICER OPERATIONS
			C_{SN} - SERVICER DDT&E
	a - LAUNCH SHARING FACTOR FOR ORBITER		b - LAUNCH SHARING FACTOR FOR TUG
	C_o - ORBITER LAUNCH COST		C_t - TUG LAUNCH COST
	λ_o - LOAD FACTOR, ORBITER		λ_t - LOAD FACTOR, TUG

B. MISSION MODEL SELECTION

For the cost analysis, the expendable mode consisted of launching the desired number of satellites to obtain the desired on-orbit fleet size and then replacing each satellite as it fails until the program is complete. The ground-refurbishable maintenance mode is similar to the expendable mode except that when a satellite fails it is retrieved from space, repaired and checked out on the ground, and then again placed in orbit. When the availability requirement is high, then additional spacecraft are procured in the ground refurbishment mode so that the replacement spacecraft can be launched as soon as possible. The failed spacecraft is retrieved on the launch flight if the capability exists, otherwise on a separate flight.

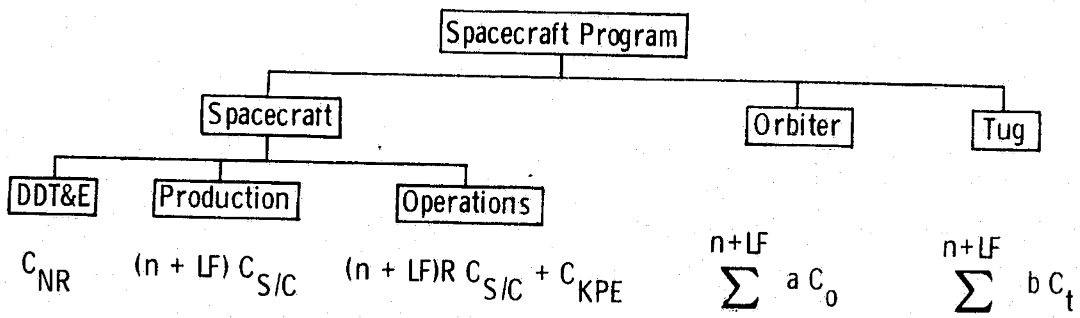

Figure IX-4 Expendable Cost Equation

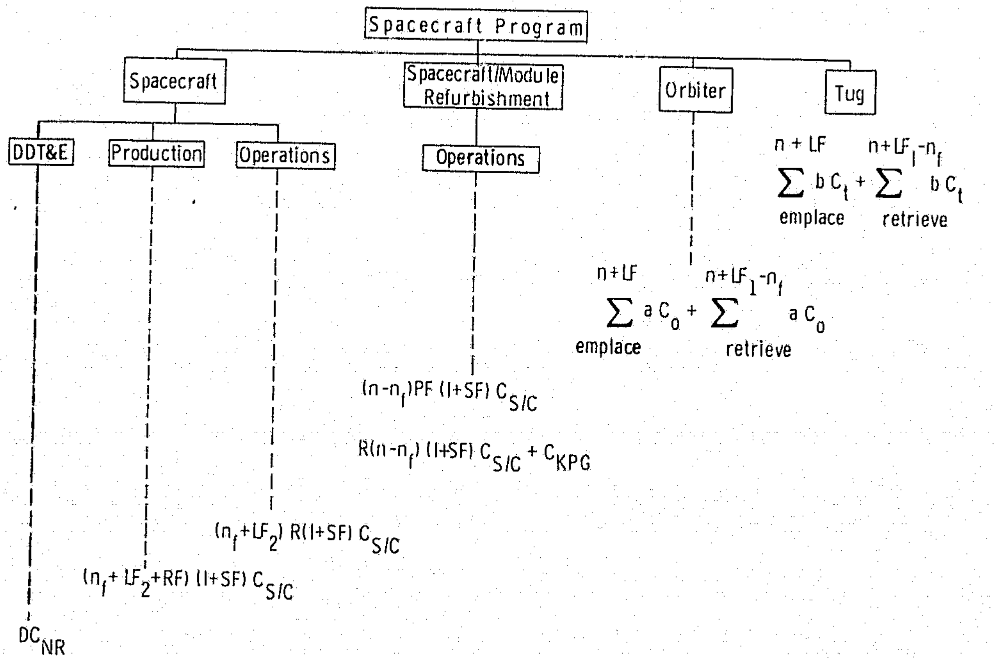

Figure IX-5 Ground Refurbishable Cost Equation

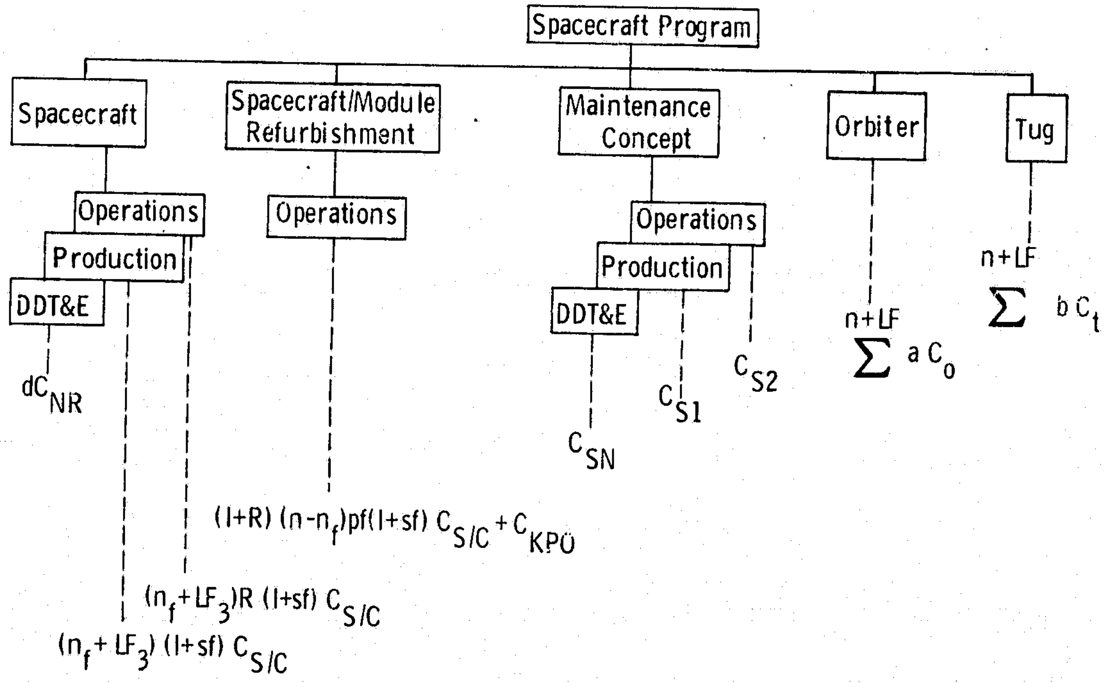

Figure IX-6 On-Orbit Maintainable Cost Equation

The on-orbit maintainable mode also starts like the expendable mode except that when a satellite fails, an on-orbit servicer carries replacement modules to the failed satellite, replaces the failed modules as well as any degraded modules, and then the on-orbit servicer returns to earth.

The economic comparison of the three maintenance modes; expendable, ground refurbishable, and on-orbit maintainable, involves a knowledge of the number of spacecraft failures expected, the number of spacecraft on-orbit at one time, the fraction of a spacecraft that should be maintained, and the number of spacecraft that might be lost. Several of the prior and current studies have obtained measures of these items by use of reliability simulation models incorporating the Monte Carlo process. To use such a model requires input data on anticipated failure rates, or reliability functions, for the level of element (spacecraft, module, subsystem, or component) that is being considered. In our case, we were interested in module failures so that reliability parameters would have to be established at the module level. One problem that arises is that this data must be selected so that the resultant anticipated number of spacecraft failures fits the number of launches and on-orbit fleet sizes of the SSPD. This would involve an iterative process which must be performed for each of the spacecraft programs of interest. This process would involve a significant detailed amount of work and would be too refined to be consistent with the level of definition of the data in the SSPD and payload model. It was determined that the use of a simulation model in this study would have been too precise with regard to the proper sensitivity for the decisions to be made.

The decision on which maintenance mode to develop should not be based on the assumed reliability of some module in some spacecraft; reliability estimates can change and certain spacecraft programs might never exist.

More explicity, one of the objectives of our study was to select and justify between the three maintenance mode alternatives: expendable, ground refurbishable, and on-orbit maintainable. Part of that selection involved an understanding of the sensitivity of the selection to the various parameters involved. Each of the spacecraft program elements would be varied about its nominal value as a part of the sensitivity study (section L) to determine its importance in evaluating the recommended mode. Therefore, it was primarily necessary that a representative number be established for each spacecraft program element. If it should develop that the mode selection critically depends on specific values of certain spacecraft elements, then a rationale different

from the comparative costing analysis would have to be used to make the decision, as we can have little confidence in any prediction of the flight situation five to twenty years in the future. (Sensitivity study results later showed that the general conclusions of the cost analysis were fairly insensitive to a wide variation in the data.)

In performing the economic evaluation, we have suggested and used an analytic method of calculating the important costing parameters rather than a Monte Carlo computer simulation model. We chose the analytic method because of the available level of input data for all the spacecraft, the ease of performing the sensitivity studies of the costing parameters input, and the proper sensitivity, with regard to the input data/output costs, as to the decision on which maintenance mode can provide the greatest economic benefits. The analytic method involved primarily the ground rule of maintaining a constant program availability and a constant spacecraft reliability across the three maintenance modes for the selected mission model. The study mission model used for costing was the maintenance applicable set of spacecraft, as described in chapter III.

This model is a very preliminary estimate and is expected to change significantly as time goes on. However, the maintenance applicable set must be considered as the best representative case on which to make the economic evaluations and the data in it can be accepted. It is possible to accept some uncertainty in the spacecraft program elements. The effects of the uncertainties were fully evaluated in the sensitivity study.

Before that mission model could be used for the sensitivity study, it was necessary to eliminate inconsistencies in the data which could project errors or uncertainties into the cost comparison. The basic premise of the analytic technique is that the mission model presents a basic reliability of a spacecraft and a basic availability for the spacecraft program and these must be held constant across the three modes. The three parameters which represent that data are the number of operating cycles (n), the on-orbit fleet size (n_f), and the average operating time (AOT) of each spacecraft.

For example, if the performance of a certain group of experiments requires one spacecraft in orbit and the mission model shows one launch per year for 5 years, then it can be assumed that that spacecraft is

expected to perform reliably for one year after launch before an expected failure, and that a replacement spacecraft would be launched after one year, and each succeeding year, up to four replacement flights. Even though that spacecraft may eventually be developed and built with a higher reliability, we must use an expendable spacecraft with a one year average operating time (AOT) with four replacement flights as the basis for costing across the expendable, ground refurbishable, and on-orbit maintenance modes. We could then accomplish that mission with five expendable spacecraft, with one spacecraft that is brought back to the ground and refurbished once each year, or with one spacecraft that is refurbished on-orbit each year. This case would be represented by the number of operating cycles being five ($n = 5$), the on-orbit fleet size of one ($n_f = 1$) and the average operating time being 1 year (AOT = 1).

Similarly, where the SSPD shows a spacecraft that is launched and then serviced on-orbit every year for four years after the initial launch, it is postulated that this mission can also be performed by launching five one-year expendable spacecraft, or by returning it to the ground and repairing it each year.

These basic postulates on spacecraft reliability are necessary to insure that the costing of three separate maintenance modes was performed equitably.

The raw data in the mission model presented these parameters, but in many cases the data was inconsistent and had to be revised. For example, the SSPD launch schedule for INTELSAT showed 16 launches, an on-orbit fleet size of between 7 and 10, and an average operational time of from two to 16 years. In another source, the average operational time was shown as ≥ 1. These types of inconsistencies existed for about half of the spacecraft. The method used to reconcile these inconsistencies involved the following general steps:

1) Obtain correct on-orbit fleet size for each spacecraft program from an independent source or analysis of mission objective.
2) Maintain the same number of flights as in the mission model.
3) Maintain the same launch schedule as in mission model.

C-4

4) Modify average operational time to be compatible with correct on-orbit fleet size.

5) Modify launch schedule as required if step 4) is not sufficient.

The original maintenance applicable set contained 49 spacecraft programs with 335 missions. A revised mission model for the maintenance applicable set of 47 spacecraft programs and 317 missions was established for the preliminary cost analysis. During the performance of the preliminary analysis, it was discovered that the mission model being used was penalizing the on-orbit maintainable and ground refurbishable modes for some programs during the later portion of the period being investigated in that spacecraft were being launched without being serviced. Since the cost benefits of maintenance, whether on the ground or in-orbit, occurs only when a spacecraft has been serviced, and a cost penalty must be paid to first produce and launch a maintainable spacecraft, it was felt that to make the cost comparison between the three modes more equitable, it was necessary to complete some amount of servicing on all spacecraft that were launched. This made it necessary to consider missions past 1991. As a minimum, it was decided to service each spacecraft that was launched prior to 1991 at least once (that would be compared with launching another expendable spacecraft at the same time). Since the preliminary cost analysis also showed that the more a spacecraft was serviced, the greater the savings over the expendable mode, it was also decided to extend certain spacecraft programs in the period after 1991 by having more than one servicing per spacecraft. This was done on spacecraft programs started immediately prior to 1991 that were planned to last for longer than the 2 or 3 year period until 1991. The revised mission model was then expanded for the 47 programs from 317 missions to 340 missions, all 23 additional missions occurring after 1991. Figure IX-7 presents the maintenance mission model used for the final cost analysis. Table IX-4 presents a summary of the maintenance applicable set mission model data for values of n, n_f, and AOT as used during the various phases of this study. The "ORIGINAL" column lists n, n_f, and AOT values originally developed during the first half of the study. The "PRELIMINARY" column lists the n, n_f, and AOT values for the revised mission model used during the preliminary cost analysis in the third quarter of this study. The "FINAL" column lists the final mission

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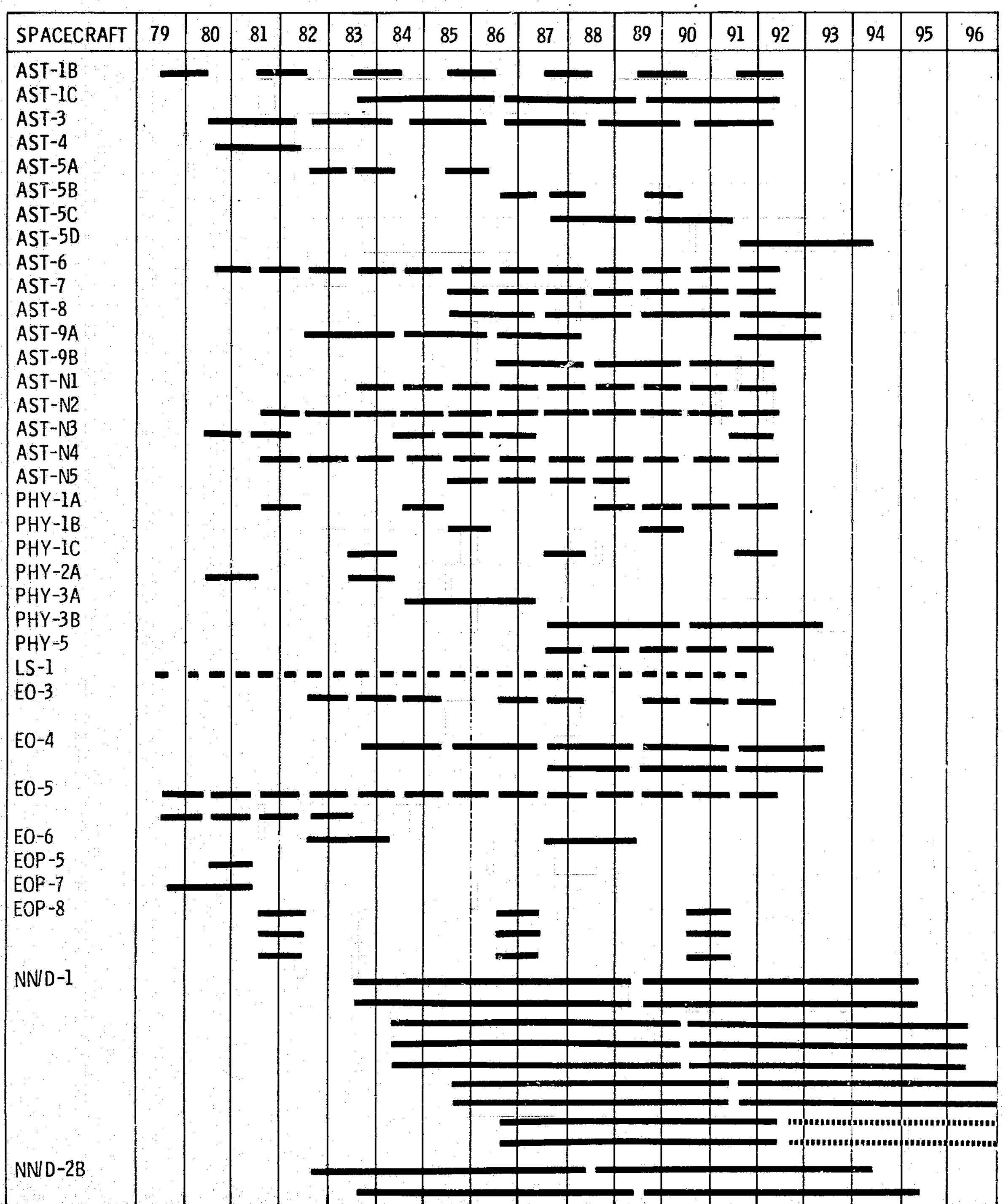

Figure IX-7 Mission Model Launch Schedule

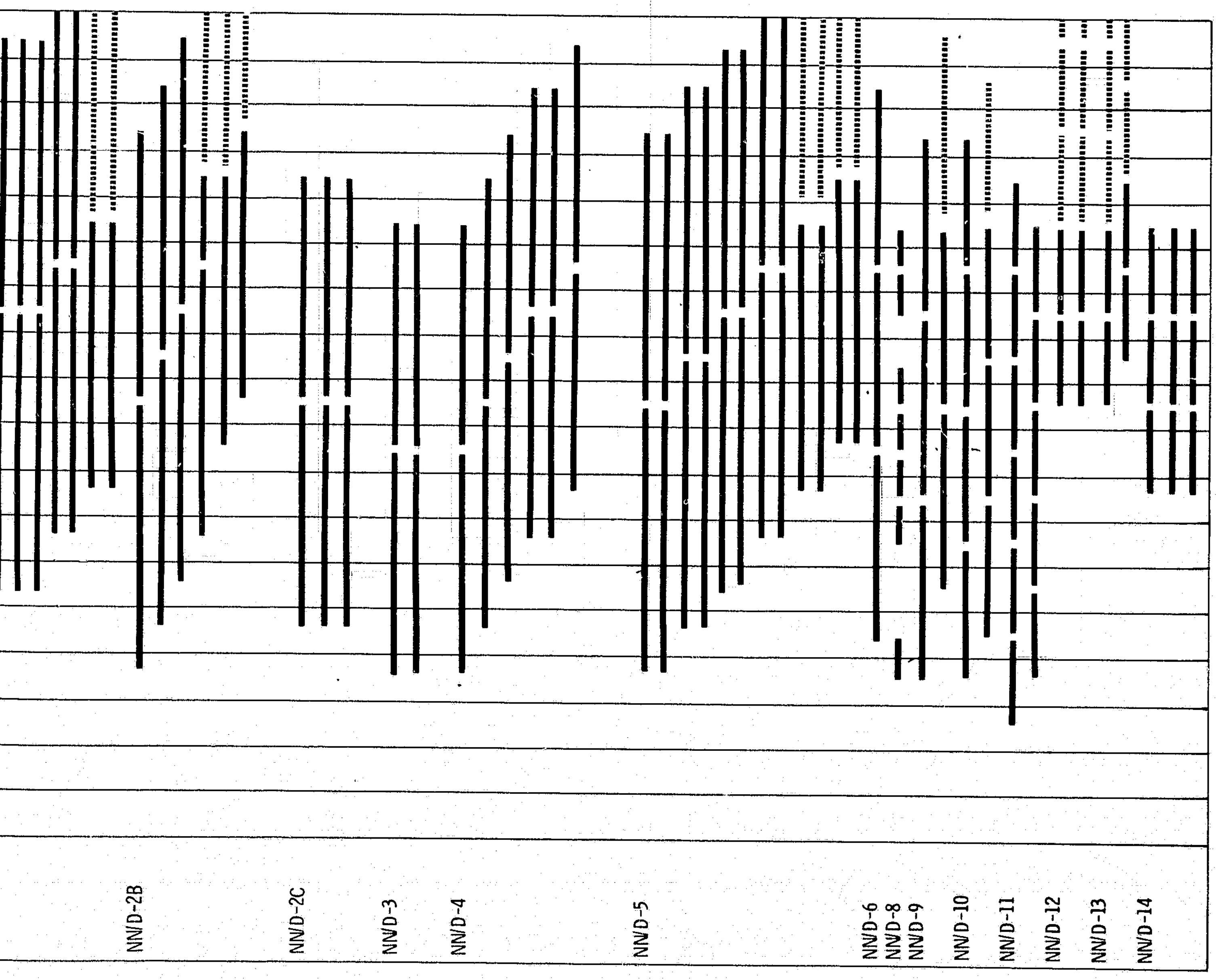

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Table IX-4 Maintenance Mission Model Summary

Payload No.	Payload Model Code No.	LEO Spacecraft Name	ORIGINAL n	ORIGINAL n_f	ORIGINAL AOT	PRELIMINARY n	PRELIMINARY n_f	PRELIMINARY AOT	FINAL n	FINAL n_f	FINAL AOT
AS-03-A	AST-1B	Cosmic Background Explorer	7	1	1	7	1	1	7	1	1
SO-03-A	AST-3	Solar Maximum Mission	6	1	2	6	1	2	6	1	2
HE-09-A	AST-4	Large High Energy Observatory B	2	1	2	1	1	2	1	1	2
HE-03-A	AST-5A	Extended X-Ray Survey	3	1-2	1	3	1	1	3	1	1
HE-08-A	AST-5B	Large High Energy Observatory A	3	1-2	1	3	1	1	3	1	1
HE-10-A	AST-5C	Large High Energy Observatory C	2	1	2	2	1	2	2	1	2
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	1	1	3	1	1	3	1	1	3
AS-01-A	AST-6	Large Space Telescope	12	1	1	12	1	1	12	1	1
SO-02-A	AST-7	Large Solar Observatory	7*	1-2	1	7	1	1	7	1	1
HE-11-A	AST-9A	Large High Energy Observatory D	4*	1-2	1	4	1	2	4	1	2
HE-01-A	AST-9B	Large X-Ray Telescope Facility	3	1-2	1	3	1	2	3	1	2
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	9	1	1	9	1	1	9	1	1
AS-11-A	AST-N2	1.5m IR Telescope	36	1	.25	11	1	1	11	1	1
AS-13-A	AST-N3	UV Survey Telescope	6	1	.25	6	1	1	6	1	1
AS-14-A	AST-N4	1m UV - Optical Telescope	11	1	1	11	1	1	11	1	1
AS-17-A	AST-N5	30m IR Interferometer	4	1	1	4	1	1	4	1	1
HE-07-A	PHY-1A	Small High Energy Satellite	6	1	1	6	1	1	6	1	1
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	2	1	1	2	1	1	2	1	1
HE-12-A	PHY-5	Cosmic Ray Laboratory	5	1	1	5	1	1	5	1	1
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	25	1	.5	25	1	.5	25	1	.5
EO-08-A	EO-3	Earth Observatory Satellite	19	1-4	1-2	16	2	1	16	2	1
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	16	1	1-2	16	2	1	16	2	1
OP-02-A	EOP-5	Gravity Gradiometer	1	1	1	1	1	1	1	1	1
OP-04-A	EOP-7	GRAVSAT	1	1	2	1	1	2	1	1	2
OP-05-A	EOP-8	Vector Magnetometer Satellite	9	3	1	9	3	1	9	3	1
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	9	3	2	9	3	2	9	3	2
		TOTAL LEO	209	30-38	-	180	32	-	180	32	-
		MEO/HEO Spacecraft Name									
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	3	1-2	3	3	1	3	3	1	3
AS-16-A	AST-8	Large Radio Observatory Array	4	1	2	4	1	2	4	1	2
AP-01-A	PHY-1B	Upper Atmosphere Explorer	2	1	1	2	1	1	2	1	1
AP-02-A	PHY-1C	Explorer-Medium Altitude	3	1	1	3	1	1	3	1	1
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	1	1	3	1	1	3	1	1	3
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	2	1	3	2	1	3	2	1	3
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	8	2	1-2	8	2	2	8	2	2
EO-12-A	EO-6	TIROS	2	1	2	2	1	2	2	1	2
CN-51-A	NN/D-1	INTELSAT	16	7-10	2-16	16	9	6	18	9	6
CN-53-A	NN/D-2B	DOMSAT B	14	10	7-14	11	7	6	14	7	6
CN-58-A	NN/D-2C	DOMSAT C	6	5-7	3-6	6	3	5	6	3	5
CN-54-A	NN/D-3	Disaster Warning Satellite	3	5	2-4	4	2	5	4	2	5
CN-55-A	NN/D-4	Traffic Management Satellite	6	5	2-7	14	7	5	14	7	5
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	7	7	1-3	20	12	6	24	12	6
CN-59-A	NN/D-6	Communications R&D Prototype	3	3-5	1-2	3	1	4	3	1	4
EO-56-A	NN/D-8	Environmental Monitoring Satellite	7	1-2	1-2	7	1	1	7	1	1
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	5	4-5	1-3	5	2	4	6	2	4
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	7	5	3-4	7	2	3	8	2	3
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	13	1-2	1-2	11	2	2	11	2	2
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	4	2	2	4	2	2	10	2	2
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	4	1-2	1-3	4	2	2	10	2	2
CN-52-A	NN/D-2A	DOMSAT A	3	7	3-7	DROPPED FROM MAS					
CN-60-A	NN/D-5B	Foreign Communication Sat.-B	3	7	4-7						
		TOTAL, MEO & HEO	126	79-91	-	137	61	-	160	61	-
		TOTAL	335	109-129	-	317	93	-	340	93	-

model used in the final cost analyses and includes the 23 additional missions flown after 1991 to provide a more equitable comparison between expendable and maintainable modes.

Another factor observed during the performance of the preliminary cost analysis indicated that the mission model used also could penalize some expendable spacecraft programs during the early portion of the shuttle era. This was because in the cost analysis a total DDT&E cost was included for each expendable program, and for each maintainable program, regardless of whether or not those programs had flown expendable spacecraft prior to 1979. If a program had flown spacecraft prior to 1979, then almost no new DDT&E would be required to fly expendable spacecraft after 1979. However, it is possible that almost all of the DDT&E budgeted for the on-orbit maintainable spacecraft would still be required, over and above the DDT&E already expended for the expendable spacecraft. It was not possible to calculate the effect of this problem and it was decided to temporarily ignore it. It was decided to designate those programs which might be affected by this problem, review the total costs at the end of the cost analysis, and then to determine the possible effects on those total program costs. Table IX-5 presents a list of the programs in the maintenance mission model which launch spacecraft prior to 1979. The 1979 date applies to low earth orbit (LEO) spacecraft only. High earth orbit spacecraft are to be flown expendable until 1982.

It should be noted here that the economic benefits of servicing are highly dependent upon the mission model being considered. The SSPD and payload model data were combined into a mission model to create as "real" a test case as possible to be used to compare costs for servicing against expendable. The actual missions flown during the shuttle era will almost certainly vary quite widely from this mission model, but it is hoped that the data in the sensitivity analyses will provide indications as to what happens to the costs as the mission model varies.

C. LAUNCH COST REIMBURSEMENT POLICY

Launch cost reimbursement policy (LCRP) is the name given to the method to be used to charge the space shuttle users and to reimburse NASA for the recurring costs of the space shuttle system including, for our purposes, both orbiter flights and tug flights. Since most of the system

Table IX-5 Spacecraft Programs with Launches Prior to 1979 (1982)

PAYLOAD NO.	PAYLOAD MODEL CODE NO.	SPACECRAFT NAME	LAUNCHES, 1979 AND AFTER	LAUNCHES PRIOR TO 1979
AS-03-A	AST-1B	COSMIC BACKGROUND EXPLORER	7	?(9 BETWEEN AST-1A, AST-1B, AST-1C)
SO-03-A	AST-3	SOLAR MAXIMUM MISSION	6	1
HE-09-A	AST-4	LHEO B	1	3
HE-07-A	PHY-1A	SMALL HIGH ENERGY SATELLITE	6	?(9 BETWEEN PHY-1A, PHY-1B, PHY-1C, PHY-1D)
EO-08-A	EO-3	EOS	16	3
EO-10-A	EO-5	APPLICATIONS EXPLORER	16	3
			LAUNCHES, 1982 AND AFTER	LAUNCHES PRIOR TO 1982
AS-05-A	AST-1C	ADVANCED RATIO AST. EXPL.	3	?(9 BETWEEN AST-1A, AST-1B, AST-1C)
AP-01-A	PHY-1B	UAE	2	?(9 BETWEEN PHY-1A, PHY-1B, PHY-1C, PHY-1D)
AP-02-A	PHY-1C	EXPLORER - MEDIUM ALTITUDE	3	?
EO-09-A	EO-4	SEOS	8	1
EO-12-A	EO-6	TIROS	2	1
CN-51-A	NN/D-1	INTELSAT	18	14
CN-53-A	NN/D-2B	DOMSAT B	14	?(20 BETWEEN NN/D-2A, NN/D-2B, MOST PROBABLY ALL ARE NN/D-2A)
CN-55-A	NN/D-4	TMS	14	11
CN-60-A	NN/D-5	FOREIGN COMSAT	24	13
EO-56-A	NN/D-8	EMS (AIR SAT)	7	2
EO-57-A	NN/D-9	FOR. SYNCH. MET. SAT.	6	2
EO-58-A	NN/D-10	GEO. OP. MET. SAT.	8	6
EO-61-A	NN/D-11	EARTH RESOURCES SURVEY OP. SAT.	11	4

will be reusable, the costs will involve mainly operations costs plus some costs for the nonreusable portion (external tank). Because one of the purposes of the STS is to provide low launch costs to potential users, a policy is required to calculate the lowest costs to users, while still permitting NASA to recoup recurring costs of the system.

The basic premise used for the LCRP in this study is that every STS user pays for the portion of the orbiter and tug capabilities that his spacecraft requires. This means that first the capabilities of the orbiter and the tug to emplace and retrieve spacecraft in various orbits must be known. Whatever percentages of this capability that a spacecraft uses is the percentage of the total launch cost that that particular spacecraft must pay.

For the orbiter, the capability is calculated in terms of the weight of the spacecraft to be carried, and the volume of the spacecraft, since it must occupy a portion of a limited volume in the cargo bay. For the tug, only the spacecraft weight is included in the capability; the spacecraft volume is important only in the cargo bay. (Weights and volumes of payload support system cradles, adapters, control and display equipment, and tug support equipment must be included, where applicable.)

The Space Shuttle Payload Accommodations document (bibliography item D-1) and the Baseline Space Tug documents (bibliography items E-5 thru E-8) provide current indications of orbiter and tug capabilities. Tables IX-6 and IX-7 and Figure IX-8 provide a summary of the orbiter and tug capabilities as used for the LCRP. Note that the orbiter volume capability is measured in terms of payload bay length utilized. This policy then tends to force a spacecraft designer to use as little of the length of the payload bay and as much of the diameter for his spacecraft as possible, resulting in potentially better space utilization of the payload bay.

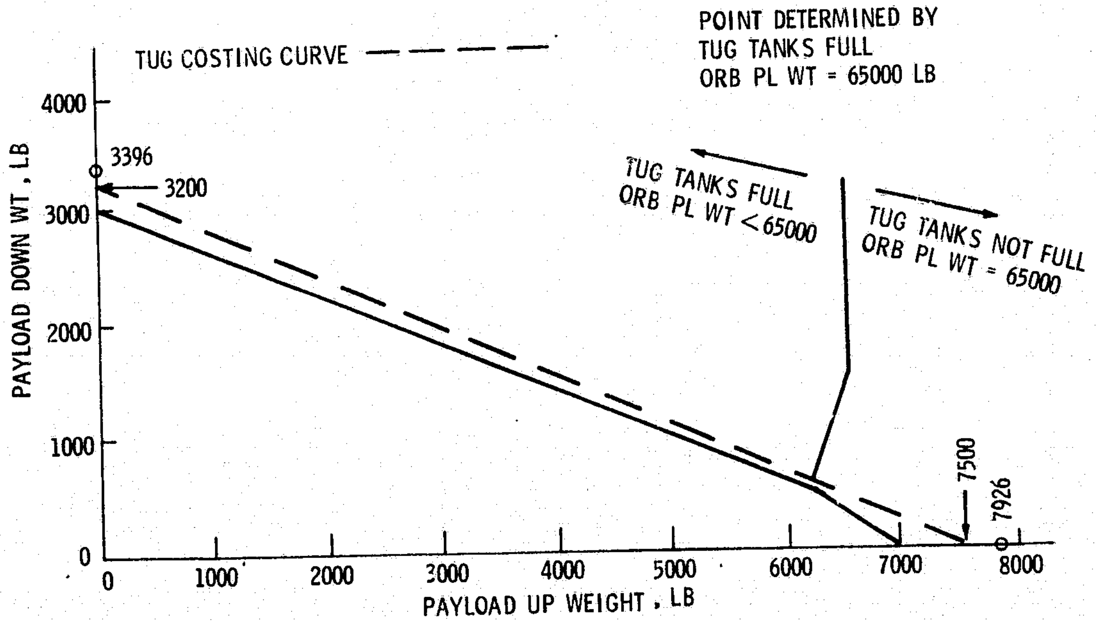

Figure IX-8 Tug Geostationary Performance

Equations were established to calculate how much weight and length each payload should be charged. Two major categories were established; low earth orbit spacecraft, delivered only by the orbiter, and tug-delivered spacecraft. For the maintenance applicable set, the tug-delivered spacecraft comprise two subgroups; those going to geostationary orbits and those going to nongeostationary orbits. These correspond, roughly, to high earth orbit (HEO) missions and medium earth orbit (MEO) missions, although one of the nongeostationary spacecraft goes to twice geosynchronous altitude but with no plane change.

Table IX-6 Orbiter and Tug Capabilities

	WEIGHT	LENGTH
ORBITER UP CAPABILITY	SEE FIGURES 3-3 THROUGH 3-12 OF BIBLIOGRAPHY ITEM D-1 65,000 lb MAXIMUM, ALSO NOMINAL FOR 28.5° INCLINATION, UP TO 190 n mi ORBIT	60 ft WITHOUT OMS KITS 50.3 ft WITH 1, 2, or 3 OMS KITS
ORBITER DOWN CAPABILITY	SEE FIGURE 3-15 OF BIBLIOGRAPHY ITEM D-1 32,000 lb MAXIMUM, ALSO NOMINAL FOR ALL CIRCULAR ORBITS 250 n mi AND 90° INCLINATION	60 ft WITHOUT OMS KITS 50.3 ft WITH 1, 2, or 3 OMS KITS
TUG UP CAPABILITY	7,500 lb TO GEOSTATIONARY WITH NO DOWN (SEE FIGURE IX-8) TUG ORBITS OTHER THAN GEOSTATIONARY CALCULATED SEPARATELY. (SEE TABLE IX-8)	
TUG DOWN CAPABILITY	3,200 lb FROM GEOSTATIONARY, WITH NO UP. (SEE FIGURE IX-7) TUG ORBITS OTHER THAN GEOSTATIONARY CALCULATED SEPARATELY. (SEE TABLE IX-7)	

Table IX-7 Tug Capabilities for Nongeostationary Spacecraft

SPACECRAFT	ORBIT (APOGEE-n mi/ PERIGEE-n mi/INCLINATION - Degree)	MAX UP CAPABILITY (lb)	MAX DOWN CAPABILITY (lb)	NUMBER OF MISSIONS IN MAINTENANCE APPLICABLE SET
GEOSTATIONARY	19,323/19,323/0°	7,900	3,400	128 (80%)
AST-8	38,635/38,635/28.5°	8,850	4,130	4 (2.5%)
PHY-1B	1,900/140/90°	22,490	125,220 (ORBITER LIMITED TO 32,000)*	2 (1.25%)
PHY-1C	19,986/1,000/28.5°	23,660	25,310*	3 (1.875%)
PHY-3A	6,895/6,895/55°	16,780	16,490	1 (0.625%)
PHY-3B	6,895/6,895/55°	16,780	16,490	2 (1.25%)
EO-6	915/905/102.97°	15,510	83,910 (ORBITER LIMITED TO 32,000)*	2 (1.25%)
NND-8	915/905/102.97°	15,510	83,910 (ORBITER LIMITED TO 32,000)*	7 (4.375%)
NND-11	490/490/99°	21,620	268,720 (ORBITER LIMITED TO 32,000)*	11 (6.875%)
TOTAL				160

*TUG PLUS ORBITER INTERFACE REQUIRES 7,050 OF 32,000, LEAVING 24,950 FOR SPACECRAFT.

Table IX-8 presents a list of the equations used to calculate the weights and lengths to be used for charging, for the expendable, ground refurbishable, and on-orbit maintainable modes for LEO spacecraft, only. Table IX-9 presents the weight equations used for tug charging, and Table IX-10 presents the weight and length equations used for orbiter charging, for the MEO and HEO spacecraft. Table IX-11 provides a definition of all terms used in Tables IX-8, IX-9, and IX-10. There are some difference in the ground rules used to calculate up and down weights and lengths for Tables IX-8, IX-9, and IX-10. Table IX-12 presents a list of the ground rules used to calculate up and down weights and lengths.

Table IX-12 Ground Rules for Up and Down Weights and Lengths

FULLY-LOADED SPACECRAFT ARE LAUNCHED, WHERE APPLICABLE IN ALL THREE MODES; HOWEVER, ONLY EMPTY SPACECRAFT (CONSUMABLES HAVE BEEN USED OR PURGED) ARE RETRIEVED FOR GROUND REFURBISHMENT.
FULLY-LOADED MODULES ARE LAUNCHED FOR ON-ORBIT MAINTENANCE; HOWEVER, EMPTY MODULES ARE RETURNED FOR LEO SPACECRAFT, WHILE NO MODULES ARE RETURNED FOR TUG-DELIVERED SPACECRAFT.
THE PAYLOAD SUPPORT EQUIPMENT IS RETRIEVED FOR ALL MODES FOR LEO SPACECRAFT; THE PAYLOAD SUPPORT EQUIPMENT IS NOT RETURNED FOR ALL MODES FOR TUG-DELIVERED SPACECRAFT.
THE B FACTOR IS MADE UP OF B UP AND B DOWN. WHERE B DOWN IS NOT ZERO, B UP IS USED TO CALCULATE ORBITER UP CHARGES AND B DOWN IS USED TO CALCULATE ORBITER DOWN CHARGES. WHERE B DOWN IS ZERO, B UP IS USED FOR BOTH.

After the calculation of the STS capabilities, and the calculation of the up and down weights and lengths, the next step is to combine them into the correct terms for the LCRP. Table IX-13 presents the equations used to do this. Note the basic difference between the equations used for orbiter and for tug. The reason they are different is that the up capability and the down capability of the orbiter are indepedent of each other, while on the tug, they are directly dependent on each other, as was shown in Figure IX-8. For the orbiter, $a_{up} = Mx\left[f_1, f_2\right]$, $a_{down} =$

Table IX-8 Weight and Length Equations for LEO Spacecraft

EXPENDABLE

* $W_{UP} = W_{S/C} + W_{C\&D} + W_{PL\ SUP}$	(for n)
$W_{DN} = W_{C\&D} + W_{PL\ SUP}$	(for n)
$L_{UP} = L_{S/C} + L_{PL\ SUP}$	(for n)
$L_{DN} = L_{PL\ SUP}$	(for n)

GROUND REFURBISHABLE

$W_{UP} = W_{S/C} + W_{C\&D} + W_{PL\ SUP}$	(for n)
$W_{DN} = W_{S/C} - W_{CONS} + W_{C\&D} + W_{PL\ SUP}$	(for $n-n_f$)
$W_{DN} = W_{C\&D} + W_{PL\ SUP}$	(for n_f)
$L_{UP} = L_{S/C} + L_{PL\ SUP}$	(for n)
$L_{DN} = L_{S/C} + L_{PL\ SUP}$	(for $n-n_f$)
$L_{DN} = L_{PL\ SUP}$	(for n_f)

ON-ORBIT MAINTAINABLE

$W_{UP} = W_{S/C} + W_{C\&D} + W_{PL\ SUP}$	(for n_f)
$W_{DN} = W_{C\&D} + W_{PL\ SUP}$	(for n_f)
$W_{UP} = W_{SER} + W_{MODU} + W_{C\&D}$	(for $n-n_f$)
$W_{DN} = W_{SER} + W_{MODD} + W_{C\&D}$	(for $n-n_f$)
$L_{UP} = L_{S/C} + L_{PL\ SUP}$	(for n_f)
$L_{DN} = L_{PL\ SUP}$	(for n_f)
$L_{UP} = L_{SER}$	(for $n-n_f$)
$L_{DN} = L_{SER}$	(for $n-n_f$)

* SEE TABLE IX-11 FOR DEFINITION OF TERMS

Table IX-9 Weight Equations for Tug for MEO & HEO Spacecraft

EXPENDABLE

$* \; W_{UP} = W_{S/C} + W_{PL\ SUP}$ (for n)

GROUND REFURBISHABLE

$W_{UP} = W_{S/C} + W_{PL\ SUP}$ (for n)

$W_{DN} = W_{S/C} - W_{CONS} + W_{PL\ SUP}$ (for $n-n_f$)

ON-ORBIT MAINTAINABLE

$W_{UP} = W_{S/C} + W_{PL\ SUP}$ (for n_f)

$W_{UP} = W_{SER} + W_{MODU}$ (for $n-n_f$)

$W_{DN} = W_{SER}$ (for $n-n_f$)

* SEE TABLE IX-11 FOR DEFINITION OF TERMS

Table IX-10 Weight and Length Equations for Orbiter for MEO & HEO Spacecraft

EXPENDABLE

$*W_{UP} = W_{S/C} + W_{PL\ SUP} + W_{C\&D} + b(W_{TUG\ F})$ (for n)

$W_{DN} = W_{C\&D} + b(W_{TUG\ E})$ (for n)

$L_{UP} = L_{S/C} + L_{PL\ SUP} + b(L_{TUG})$ (for n)

$L_{DN} = b\ (L_{TUG})$

GROUND REFURBISHABLE

$W_{UP} = W_{S/C} + W_{C\&D} + W_{PL\ SUP} + b(W_{TUG\ F})$ (for n)

$W_{DN} = W_{S/C} - W_{CONS} + W_{PL\ SUP} + W_{C\&D} + b(W_{TUG\ E})$ (for $n-n_f$)

$W_{DN} = W_{C\&D} + b(W_{TUG\ E})$ (for n_f)

$L_{UP} = L_{S/C} + L_{PL\ SUP} + b(L_{TUG})$ (for n)

$L_{DN} = L_{S/C} + L_{PL\ SUP} + b(L_{TUG})$ (for $n-n_f$)

$L_{DN} = b(L_{TUG})$ (for n_f)

ON-ORBIT MAINTAINABLE

$W_{UP} = W_{S/C} + W_{PL\ SUP} + W_{C\&D} + b(W_{TUG\ F})$ (for n_f)

$W_{DN} = W_{C\&D} + b(W_{TUG\ E})$ (for n_f)

$W_{UP} = W_{SER} + W_{MODU} + W_{C\&D} + b(W_{TUG\ F})$ (for $n-n_f$)

$W_{DN} = W_{SER} + W_{C\&D} + b(W_{TUG\ E})$ (for $n-n_f$)

$L_{UP} = L_{S/C} + L_{PL\ SUP} + b(L_{TUG})$ (for n_f)

$L_{DN} = b\ (L_{TUG})$ (for n_f)

$L_{UP} = L_{SERV} + b(L_{TUG})$ (for $n-n_f$)

$L_{DN} = L_{SERV} + b(L_{TUG})$ (for $n-n_f$)

* SEE TABLE IX-11 FOR DEFINITION OF TERMS

Table IX-11 Definition of Terms

W_{UP}	=	Weight carried up, charged to particular program, lbs
W_{DN}	=	Weight carried down, charged to particular program, lbs
L_{UP}	=	Length carried up, charged to particular program, ft
L_{DN}	=	Length carried down, charged to particular program, ft
$W_{S/C}$	=	Fully-loaded spacecraft weight, lbs
$W_{C\&D}$	=	Weight of spacecraft-dedicated controls and displays, lbs
$W_{PL\ SUP}$	=	Payload bay spacecraft support equipment weight, lbs
$L_{S/C}$	=	Length of spacecraft, ft
$L_{PL\ SUP}$	=	Payload bay spacecraft support equipment length, ft
W_{CONS}	=	Weight of consumables in spacecraft, lbs
W_{SER}	=	Servicer weight, lbs
W_{MODU}	=	Weight of modules carried up (fully loaded), lbs
W_{MODD}	=	Weight of modules carried down (empty), lbs
L_{SER}	=	Servicer length, ft
$W_{TUG\ F}$	=	Weight of fully loaded tug in payload bay, lbs
$W_{TUG\ E}$	=	Weight of empty tug in payload bay, lbs
L_{TUG}	=	Length of tug, ft
b	=	Tug charging factor used to charge tug up and down to orbiter costs

Table IX-13 Shuttle User Charges - Launch Cost Reimbursement Policy Evaluations

CHARGE	ORBITER	TUG
PL UP	$\mathrm{Mx}\left[f_1, f_2\right] \dfrac{C\ orb}{\lambda_{ORB}}$	$f_5 \dfrac{C\ Tug}{\lambda_{TUG}}$
PL DOWN	$\mathrm{Mx}\left[f_3, f_4\right] \dfrac{C\ orb}{\lambda_{ORB}}$	$f_6 \dfrac{C\ Tug}{\lambda_{TUG}}$

$$f_1 = \frac{\text{PL up wt}}{\text{PL up wt capability + PL dn wt capability}}$$

$$f_2 = \frac{\text{PL up length}}{\text{PL up length capability + PL dn length capability}}$$

$$f_3 = \frac{\text{PL dn wt}}{\text{PL up wt capability + PL dn wt capability}}$$

$$f_4 = \frac{\text{PL dn length}}{\text{PL up length capability + PL dn length capability}}$$

$$f_5 = \frac{\text{PL up wt}}{\text{PL up wt capability}}$$

$$f_6 = \frac{\text{PL dn wt}}{\text{PL dn wt capability}}$$

$Mx\left[f_3,f_4\right]$ and $a = a_{up} + a_{down} = Mx\left[f_1,f_2\right] + Mx\left[f_3,f_4\right]$. For the tug, $b_{up} = f_5$, $b_{down} = f_6$, and $b = b_{up} + b_{down} = f_5 + f_6$. The average load factors for the orbiter and the tug, $\lambda_{orb} = 0.70$ and $\lambda_{tug} = 0.85$, were suggested by NASA. Table IX-14 presents a summary of the spacecraft in the maintenance applicable set and the values used to calculate up and down weights and lengths. These values were taken mostly from the SSPD and the payload model and modified for use, as appropriate, although data on the payload support equipment weights and lengths were based on data from the MMC Study, Multi-Use Mission Support Equipment, NAS8-30847 (bibliography items J-10, -11, and -12). Table IX-15 presents the up and down capabilities as used for the maintenance applicable set, and Table IX-16 presents a summary of a and b for the spacecraft in the maintenance applicable set.

Table IX-17 presents another form of summary of the data from the launch cost reimbursement policy as used for this program. The data is presented in the form of "equivalent orbiter flights", and "equivalent tug flights" required to fly the maintenance applicable set in the expendable, ground refurbishable, and on-orbit maintainable modes. As can be seen, the number of "equivalent orbiters" required for the on-orbit maintainable mode is quite a bit less than for expendable, while the number of "equivalent tugs" is about the same. The numbers of "equivalent orbiters" and of "equivalent tugs" can be converted directly into cost. (The ground refurbishable mode clearly requires more tug and orbiter flights.)

One of the purposes of utilizing a form of LCRP to cost the maintenance modes was to show the dependence of results on the form of the LCRP used. The value of total savings and costs, the selections of the most economic maintenance mode, and maintenance operations considerations (return module or not, expendable servicer, etc.) all depend on the form of the LCRP used. The variations in the parameters used in this study for the LCRP were investigated in the cost sensitivity study. However, changes in the methods used to calculate LCRP as suggested in Table IX-18 can have an effect on the results of this study.

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Table IX-14 Up and Down Weights and Lengths Summary

Payload No.	Payload Model Code No.	LEO Spacecraft Name	$W_{S/C}$ EX	$W_{S/C}$ GR	$W_{S/C}$ OOM	$W_{C\&D}$	W_{PL} S
AS-03-A	AST-1B	Cosmic Background Explorer	1312	1347	1980	200	900
SO-03-A	AST-3	Solar Maximum Mission	1643	1678	2311	26	
HE-09-A	AST-4	Large High Energy Observatory B	13792	13827	14460	265	
HE-03-A	AST-5A	Extended X-Ray Survey	16996	17031	17664	265	
HE-08-A	AST-5B	Large High Energy Observatory A	18515	18550	19183	265	
HE-10-A	AST-5C	Large High Energy Observatory C	11098	11133	11766	265	
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	14904	14939	15572	265	
AS-01-A	AST-6	Large Space Telescope	20169	20204	20837	265	
SO-02-A	AST-7	Large Solar Observatory	20996	21031	21664	1153	
HE-11-A	AST-9A	Large High Energy Observatory D	14262	14297	14930	265	
HE-01-A	AST-9B	Large X-Ray Telescope Facility	21209	21244	21877	265	
AS-07-A	AST-N1	3m Ambient Temperature IR Télescope	18282	18317	18950	265	
AS-11-A	AST-N2	1.5m IR Telescope	12661	12696	13329	265	
AS-13-A	AST-N3	UV Survey Telescope	7528	7563	8196	265	
AS-14-A	AST-N4	1m UV - Optical Telescope	8254	8289	8922	265	
AS-17-A	AST-N5	30m IR Interferometer	7027	7062	7695	265	
HE-07-A	PHY-1A	Small High Energy Satellite	1311	1346	1979	200	
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	1323	1358	1991	251	
HE-12-A	PHY-5	Cosmic Ray Laboratory	21453	21488	22121	265	
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	4965	5000	5633	265	
EO-08-A	EO-3	Earth Observatory Satellite	6994	7029	7662	100	
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	310	345	978	100	
OP-02-A	EOP-5	Gravity Gradiometer	7226	7210	7843	100	
OP-04-A	EOP-7	GRAVSAT	6805	6840	8141	100	
OP-05-A	EOP-8	Vector Magnetometer Satellite	310	345	978	100	
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	2841	2876	3509	100	
		MEO/HEO Spacecraft Name					
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	2644	2714	3980	232	45
AS-16-A	AST-8	Large Radio Observatory Array	2199	2234	2867	265	
AP-01-A	PHY-1B	Upper Atmosphere Explorer	2004	1988	2621	251	
AP-02-A	PHY-1C	Explorer-Medium Altitude	599	583	1216	251	
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	3281	3316	3949	251	
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	8701	8736	9369	551	
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	3376	3411	4044	100	
EO-12-A	EO-6	TIROS	4706	4741	5374	100	
CN-51-A	NN/D-1	INTELSAT	2090	2125	2513	100	
CN-53-A	NN/D-2B	DOMSAT B	3246	3281	3914	100	
CN-58-A	NN/D-2C	DOMSAT C	1913	1948	2581	100	
CN-54-A	NN/D-3	Disaster Warning Satellite	1285	1320	1953	100	
CN-55-A	NN/D-4	Traffic Management Satellite	658	693	1326	100	
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	705	740	1373	100	
CN-59-A	NN/D-6	Communications R&D Prototype	2109	2144	2777	100	
EO-56-A	NN/D-8	Environmental Monitoring Satellite	4860	4895	5528	100	
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	566	550	1183	100	
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	566	550	1183	100	
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	1616	1651	2284	100	
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	3376	3411	4044	100	
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	3376	3411	4044	100	

FOLDOUT FRAME

	$W_{C\&D}$	$W_{PL\ SUP}$	$L_{S/C}$	$L_{PL\ SUP}$	W_{CONS}	W_{SER}	W_{MODU}	W_{MODD}	L_{SER}			
	200	900	10.3	7.0	250	1030	558	495	4.0			
	26		10.5	7.0	65		404	393				
	265		30.0	2.0	2760		1550	1301				
	265		30.0	2.0	1222		1700	1590				
	265		30.0	2.0	1266		1840	1726				
	265		30.0	2.0	1222		1169	1059				
	265		30.0	2.0	1222		1511	1401				
	265		42.3	2.0	2020		2057	1875				
	1153		58.1	2.0	2103		2139	1950				
	265		33.5	2.0	1222		1454	1344				
	265		52.5	2.0	1466		1401	1313				
	265		40.1	2.0	3647		2034	1706				
	265		30.0	2.0	2684		1441	1200				
	265		30.0	2.0	220		757	738				
	265		30.0	2.0	1222		913	803				
	265		55.9	2.0	265		716	693				
	200		8.6	7.0	257		201	178				
	251		11.7	7.0	298		275	239				
	265		30.0	2.0	1492		2121	1991				
	265		20.0	7.0	1500		713	563				
	100		36.1	7.0	1150		1410	1226				
3	100		3.5	7.0	10		287	284				
3	100		15.1	7.0	1868		874	706				
	100		8.9	7.0	574		2092	1954				
3	100		4.0	7.0	41		285	274				
9	100		12.0	7.0	288		378	351				
										$W_{TUG\ F}$	$W_{TUG\ E}$	L_{TUG}
0	232	450	8.1	2.0	248	580	930		4.0	58879	7050	30
7	265		17.0		220		587					
1	251		7.8		386		271					
6	251		6.0		4		98					
9	251		12.1		261		884					
9	551		15.1		874		1536					
4	100		17.1		146		629					
4	100		13.3		27		972					
3	100		8.2		365		806					
4	100		8.2		120		1130					
1	100		12.1		174		854					
3	100		16.8		25		475					
6	100		10.6		39		396					
73	100		7.7		154		366					
77	100		19.7		168		736					
28	100		12.2		27		889					
33	100		10.3		35		451					
33	100		10.3		35		463					
34	100		10.0		32		301					
44	100		17.1		146		629					
44	100		17.1		146		587					

FOLDOUT FRAME 2

Table IX-15 Orbiter and Tug Up and Down Weight and Length Capabilities

Payload No.	Payload Model Code No.	LEO Spacecraft Name	Orb Up Wt	Orb Up Lng	Orb Dn Wt	Orb Dn Lng	Tug Up Wt	Tug Dn Wt
AS-03-A	AST-1B	Cosmic Background Explorer	62K	60	32K	60		
SO-03-A	AST-3	Solar Maximum Mission	52K	0.3	29.2K	50.3		
HE-09-A	AST-4	Large High Energy Observatory B	64K	60	32K	60		
HE-03-A	AST-5A	Extended X-Ray Survey	64K	60	32K	60		
HE-08-A	AST-5B	Large High Energy Observatory A	64K	60	32K	60		
HE-10-A	AST-5C	Large High Energy Observatory C	64K	60	32K	60		
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	52K	60	32K	60		
AS-01-A	AST-6	Large Space Telescope	52K	50.3	29.2K	50.3		
SO-02-A	AST-7	Large Solar Observatory	65K	60	32K	60		
HE-11-A	AST-9A	Large High Energy Observatory D	56K	50.3	29.2K	50.3		
HE-01-A	AST-9B	Large X-Ray Telescope Facility	56K	50.3	29.2K	50.3		
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	62K	60	32K	60		
AS-11-A	AST-N2	1.5m IR Telescope	62K	60	32K	60		
AS-13-A	AST-N3	UV Survey Telescope	65K	60	32K	60		
AS-14-A	AST-N4	1m UV - Optical Telescope	65K	60	32K	60		
AS-17-A	AST-N5	30m IR Interferometer	40K	50.3	28.3K	50.3		
HE-07-A	PHY-1A	Small High Energy Satellite	64K	60	32K	60		
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	21K	50.3	27.2K	50.3		
HE-12-A	PHY-5	Cosmic Ray Laboratory	64K	60	32K	60		
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	49K	50.3	29.2K	50.3		
EO-08-A	EO-3	Earth Observatory Satellite	8K	50.3	10K	50.3		
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	14K	50.3	15.2K	50.3		
OP-02-A	EOP-5	Gravity Gradiometer	35K	60	32K	60		
OP-04-A	EOP-7	GRAVSAT	35K	60	32K	60		
OP-05-A	EOP-8	Vector Magnetometer Satellite	28K	50.3	29.2K	50.3		
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	25K	50.3	22.2K	50.3		
		MEO/HEO Spacecraft Name						
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	65K	60	32K	60	7500	3200
AS-16-A	AST-8	Large Radio Observatory Array	65K	60	32K	60	*	*
AP-01-A	PHY-1B	Upper Atmosphere Explorer	37.5K	60	32K	60	*	*
AP-02-A	PHY-1C	Explorer-Medium Altitude	65K	60	32K	60	*	*
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	60K	60	32K	60	*	*
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	60K	60	32K	60	*	*
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	65K	60	32K	60	7500	3200
EO-12-A	EO-6	TIROS	29.5K	60	32K	60	*	*
CN-51-A	NN/D-1	INTELSAT	65K	60	32K	60	7500	3200
CN-53-A	NN/D-2B	DOMSAT B	65K	60	32K	60	7500	3200
CN-58-A	NN/D-2C	DOMSAT C	65K	60	32K	60	7500	3200
CN-54-A	NN/D-3	Disaster Warning Satellite	65K	60	32K	60	7500	3200
CN-55-A	NN/D-4	Traffic Management Satellite	65K	60	32K	60	7500	3200
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	65K	60	32K	60	7500	3200
CN-59-A	NN/D-6	Communications R&D Prototype	65K	60	32K	60	7500	3200
EO-56-A	NN/D-8	Environmental Monitoring Satellite	29.5K	60	32K	60	*	*
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	65K	60	32K	60	7500	3200
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	65K	60	32K	60	7500	3200
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	33K	60	32K	60	*	*
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	65K	60	32K	60	7500	3200
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	65K	60	32K	60	7500	3200

*See Table 2

Wt in pounds; Lng in feet

Table IX-16 LCRP Values used for IOSS

Payload No.	Payload Model Code No.	LEO Spacecraft Name	a: EX n	a: GR n_f	a: GR $n-n_f$	a: OOM n_f	a: OOM $n-n_f$	b: EX n	b: GR n_f	b: GR $n-n_f$	b: OOM n_f	b: OOM $n-n_f$
AS-03-A	AST-1B	Cosmic Background Explorer	.20	.20	.29	.20	.07					
SO-03-A	AST-3	Solar Maximum Mission	.24	.24	.25	.24	.07					
HE-09-A	AST-4	Large High Energy Observatory B	.28	.28	.53	.28	.07					
HE-03-A	AST-5A	Extended X-Ray Survey	.28	.28	.53	.28	.07					
HE-08-A	AST-5B	Large High Energy Observatory A	.28	.28	.53	.28	.07					
HE-10-A	AST-5C	Large High Energy Observatory C	.28	.28	.53	.28	.07					
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	.28	.28	.53	.28	.07					
AS-01-A	AST-6	Large Space Telescope	.47	.46	.88	.47	.08					
SO-02-A	AST-7	Large Solar Observatory	.52	.52	1.0	.52	.09					
HE-11-A	AST-9A	Large High Energy Observatory D	.36	.37	.71	.37	.07					
HE-01-A	AST-9B	Large X-Ray Telescope Facility	.56	.56	1.0	.56	.07					
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	.37	.37	.70	.37	.07					
AS-11-A	AST-N2	1.5m IR Telescope	.28	.28	.53	.28	.07					
AS-13-A	AST-N3	UV Survey Telescope	.28	.28	.53	.28	.07					
AS-14-A	AST-N4	1m UV - Optical Telescope	.28	.28	.53	.28	.07					
AS-17-A	AST-N5	30m IR Interferometer	.60	.62	1.0	.62	.07					
HE-07-A	PHY-1A	Small High Energy Satellite	.19	.19	.26	.19	.07					
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	.26	.26	.37	.26	.07					
HE-12-A	PHY-5	Cosmic Ray Laboratory	.28	.28	.53	.28	.07					
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	.34	.34	.54	.34	.07					
EO-08-A	EO-3	Earth Observatory Satellite	.50	.50	.86	.50	.14					
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	.17	.17	.21	.17	.07					
OP-02-A	EOP-5	Gravity Gradiometer	.24	.24	.37	.24	.07					
OP-04-A	EOP-7	GRAVSAT	.19	.19	.27	.19	.09					
OP-05-A	EOP-8	Vector Magnetometer Satellite	.18	.18	.22	.18	.07					
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	.26	.26	.38	.26	.07					
		MEO/HEO Spacecraft Name										
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	.38	.38	.60	.55	.22	.41	.42	1.33	.59	.38
AS-16-A	AST-8	Large Radio Observatory Array	.76	.76	.99	.78	.78	1.0	1.0	1.0	1.0	1.0
AP-01-A	PHY-1B	Upper Atmosphere Explorer	.58	.58	.66	.58	.57	1.0	1.0	1.0	1.0	1.0
AP-02-A	PHY-1C	Explorer-Medium Altitude	.57	.57	.63	.57	.57	1.0	1.0	1.0	1.0	1.0
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	.62	.62	.73	.64	.62	1.0	1.0	1.0	1.0	1.0
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	.74	.74	1.00	.76	.64	1.0	1.0	1.0	1.0	1.0
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	.47	.47	.73	.56	.14	.51	.56	1.52	.60	.27
EO-12-A	EO-6	TIROS	.38	.38	.51	.38	.32	.50	.50	.50	.5	.5
CN-51-A	NN/D-1	INTELSAT	.32	.32	.49	.37	.20	.34	.34	1.03	.40	.37
CN-53-A	NN/D-2B	DOMSAT B	.46	.46	.68	.54	.24	.49	.5	1.50	.58	.41
CN-58-A	NN/D-2C	DOMSAT C	.30	.30	.49	.38	.21	.32	.32	1.02	.40	.37
CN-54-A	NN/D-3	Disaster Warning Satellite	.27	.27	.51	.32	.18	.23	.24	.78	.32	.32
CN-55-A	NN/D-4	Traffic Management Satellite	.19	.19	.34	.22	.17	.15	.15	.50	.24	.31
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	.16	.16	.28	.23	.16	.15	.16	.49	.24	.31
CN-59-A	NN/D-6	Communications R&D Prototype	.32	.32	.64	.40	.20	.34	.35	1.10	.43	.36
EO-56-A	NN/D-8	Environmental Monitoring Satellite	.62	.62	.74	.62	.57	1.0	1.0	1.0	1.0	1.0
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	.17	.17	.31	.21	.17	.14	.13	.44	.22	.32
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	.17	.17	.31	.21	.17	.14	.13	.44	.24	.32
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	.60	.60	.70	.60	.57	1.0	1.0	1.0	1.0	1.0
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	.47	.47	.76	.56	.19	.51	.52	.52	.60	.34
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	.47	.47	.76	.56	.19	.51	.52	.52	.60	.34

Table IX-17 Launch Cost Reimbursement Policy Summary

	n	EXPENDABLE		GROUND REFURBISHABLE		ON-ORBIT MAINTAINABLE	
		$\frac{\Sigma a}{\lambda_{ORB}}$	$\frac{\Sigma b}{\lambda_{TUG}}$	$\frac{\Sigma a}{\lambda_{ORB}}$	$\frac{\Sigma b}{\lambda_{TUG}}$	$\frac{\Sigma a}{\lambda_{ORB}}$	$\frac{\Sigma b}{\lambda_{TUG}}$
LEO	180	83.3	--	132.7	--	30.5	--
MEO & HEO	160	84.2	83.5	116.1	144.4	69.1	83.4
TOTAL	340	167.5	83.5	248.8	144.4	99.6	83.4
EQUIVALENT ORBITERS REQUIRED		168		249		100	
EQUIVALENT TUGS REQUIRED			84		145		84

NOTE: $\lambda_{ORB} = 0.70$, $\lambda_{TUG} = 0.85$

Table IX-18 Suggested Changes to LCRP

1) DIFFERENT LOAD FACTORS FOR UP AND FOR DOWN TRAFFIC
2) DIFFERENT COSTS FOR UP AND FOR DOWN TRAFFIC
3) NO COST FOR DOWN TRAFFIC
4) VARIABLE DOLLARS PER lb FOR TYPE OF MISSION, i.e., TUG, IUS, SORTIE, AUTOMATED S/C, PLANETARY, SERVICERS, ETC

D. PARTS FACTOR

The main cost savings in going from an expendable satellite program to a maintainable program is the smaller number of satellites that must be purchased. The primary item that reduces this savings is the cost of maintaining the satellites and the most important parameter in determining that cost is the parts factor, which is the ratio of the cost of the replaced modules (or refurbished parts) to the total satellite unit cost.

Parts factors represent the portion of the satellite that is repaired or replaced. The basic idea of the parts factor is expressed as:

$$\text{Parts Factor} = \frac{\text{Cost of Replaced Modules}}{\text{Satellite Unit Cost}} \approx \frac{\text{Weight of Replaced Modules}}{\text{Satellite Total Weight}}$$

The replacement modules are generally considered as modules to replace failed modules, and it was necessary to first examine the types of failures that could necessitate module replacement. Generally, three types of failures are encountered in spacecraft; design failures, random failures and wear-out failures. Design failures are failures that usually occur soon after launch and usually occur in all components of the same design in any fleet of spacecraft. Random failures can occur any time in the life of a spacecraft and are the usual type of failures handled by reliability techniques. Wear-out failures usually occur near the end of the design life of a spacecraft and represent such things as running out of propellant. The parts factors that were used in this study represented only the random failures and the wear-out failures. Design failures were handled and are discussed separately. The parts factor used here was made up of a random failure parts factor and a wear-out failure parts factor.

The module data developed by the Aerospace Corporation in the <u>Operations Analysis (Study 2.1) Payload Designs for Space Servicing</u> (see bibliography items F-19 and F-20) were used to help develop parts factors. In that study, Aerospace developed standardized modules for a large number of missions from the NASA payload model. They modularized 29 spacecraft of which 26 were represented in the 49 spacecraft programs in the maintenance applicable set. We have used the reliability data developed for their modules to help determine parts factors in our study. We did not use the fact that these modules were standardized modules. We used the Aerospace data since it represented the best available data on what typical modularized

spacecraft and modules may look like in the time period of interest for a large number of the spacecraft in the maintenance applicable set. In addition to the Aerospace data, we also used module data from bibliography items C-1, E-9, H-1 thru -13, I-1 thru -9, K-8, M-13 thru -16, and N-1 thru N-19.

The equations used to calculate parts factors are shown in Figure IX-9. As can be seen, the parts factor for ground refurbishment contains an extra term, and is discussed in more detail later. The reliabilities of the modules were determined based on the Weibull function $R(t) = \exp(-t/\alpha)^{\beta}$ where the Weibull parameters α and β were taken mainly from the Aerospace data. Wearout parts factors were calculated for such modules as propellant modules, batteries, and cryogen modules.

A study was also made to investigate the effects of performing successive servicings on the spacecraft, i.e., servicing a spacecraft for a second (or more) time which would contain some modules which were originally launched and which had one specific time period used to calculate their reliabilities, and which would contain some newer modules with a shorter time of operation used to calculate their reliabilities. The general approach used to do this is shown in Figure IX-10. In that figure, the "U's" represent the unreliabilities, which are the main element used to calculate parts factors. Let the upper part of Figure IX-10 represent the module reliability curve starting from zero time. The quantities, b, c, d, and e represent the change in unreliability from one servicing interval to the next for those modules which are never replaced. At time τ modules are replaced, on the average, to the extent of the unreliability, b. The expected reliability of these replaced modules with time can be represented as in the middle part of Figure IX-10. Note that the reliability function has been scaled by the quantity of modules involved (b) and the curve has been shifted to the right by a time interval equal to one servicing period. The unreliability at point 2τ then becomes c (from upper part of Figure IX-10) plus f (from middle part of Figure IX-10). These are replaced by new modules (c + f of them) which start their decay from unity reliability at time = 2τ. This is sketched in the lower part of Figure IX-10. The unreliability at 3τ thus becomes d + g + i and this fraction of a module

is replaced to start new at time = 3τ. Continuing in this way, the unreliability at 4τ thus becomes e + h + j + k and this portion of a module is replaced to start new at time = 4τ. The process is repeated as often as necessary to cover the servicing intervals desired. It must be done for each module (or module with a different reliability function) of each spacecraft to obtain the parts factors. The pattern of identifying the reliabilities makes their calculation quite simple.

Step 1

$$U_1 = 1 - R(\tau) = b$$

Step 2

$$c = R(\tau) - R(2\tau)$$

$$U_2 = c + f = c + bU_1$$

Step 3

$$d = R(2\tau) - R(3\tau)$$

$$U_3 = d + g + i$$
$$= d + cU_1 + bU_2$$

A computer program was set up by COMSAT Corporation and runs were made for several spacecraft. The results of those runs showed that the parts factors did not change by much at all, even up to very large numbers of successive servicings. Figure IX-11 presents some results of the successive servicings study. M is the average multiplier that was used to multiply the random failure parts factor for each spacecraft to take into account the successive servicings. The main reason M remains so close to 1 was the large number of modules with $\beta = 1$ or ≈ 1. For $\beta = 1$, a constant failure rate exists and M will equal one. The wearout parts factors are unaffected by successive servicings.

After parts factors were determined for the 26 spacecraft in the maintenance applicable set which were in the Aerospace data, the parts factors were spread to include all the remaining spacecraft in the maintenance applicable set. This was accomplished by plotting the parts factors calculated against the parameter $\frac{\text{WEIGHT}}{\text{AOT}}$ for each spacecraft as shown in Figure IX-12, and then extrapolating to the remaining spacecraft.

$$\text{PARTS FACTOR} = \frac{\text{COST OF REPLACED MODULES}}{\text{SPACECRAFT UNIT COST}} \approx \frac{\text{WEIGHT OF REPLACED MODULE}}{\text{SPACECRAFT TOTAL WEIGHT}}$$

$$pf = pf_{(RANDOM)} + pf_{(WEAROUT)} \Longrightarrow \text{ON-ORBIT MAINTAINABLE}$$

$$PF = pf_{(RANDOM)} = pf_{(WEAROUT)} + pf_{(GR)} = pf + pf_{(GR)} \Longrightarrow \text{GROUND REFURBISHABLE}$$

- $$pf_{(RANDOM)} = \sum_{i=1}^{n} \frac{(1 - R_i)\, w_i}{W_{TOT}}$$

- $$pf_{(WEAROUT)} = \sum_{i=1}^{n_w} \frac{AOT}{DL_i} \frac{w_i}{W_{TOT}}$$

- $$pf_{(GR)} = pf_{(APPENDAGES)} + pf_{\text{CLEAN, PAINT, DECONT., TEST}}$$

WHERE

R_i = RELIABILITY OF MODULE i AT AOT

W_i = WEIGHT OF MODULE i

W_{TOT} = TOTAL SPACECRAFT WEIGHT

n_w = NUMBER OF MODULES WHICH CAN HAVE WEAROUT FAILURES

DL_i = DESIGN LIFE OF MODULE i (OR OF SPACECRAFT, WHICHEVER IS LEAST)

AOT = AVERAGE OPERATING TIME IN YEARS

Figure IX-9 Parts Factors Equations

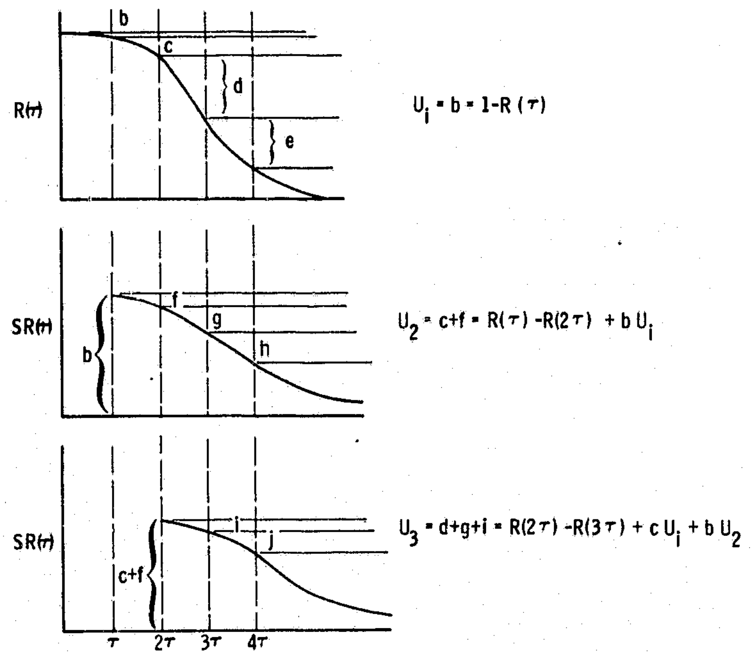

Figure IX-10 Successive Servicing Method

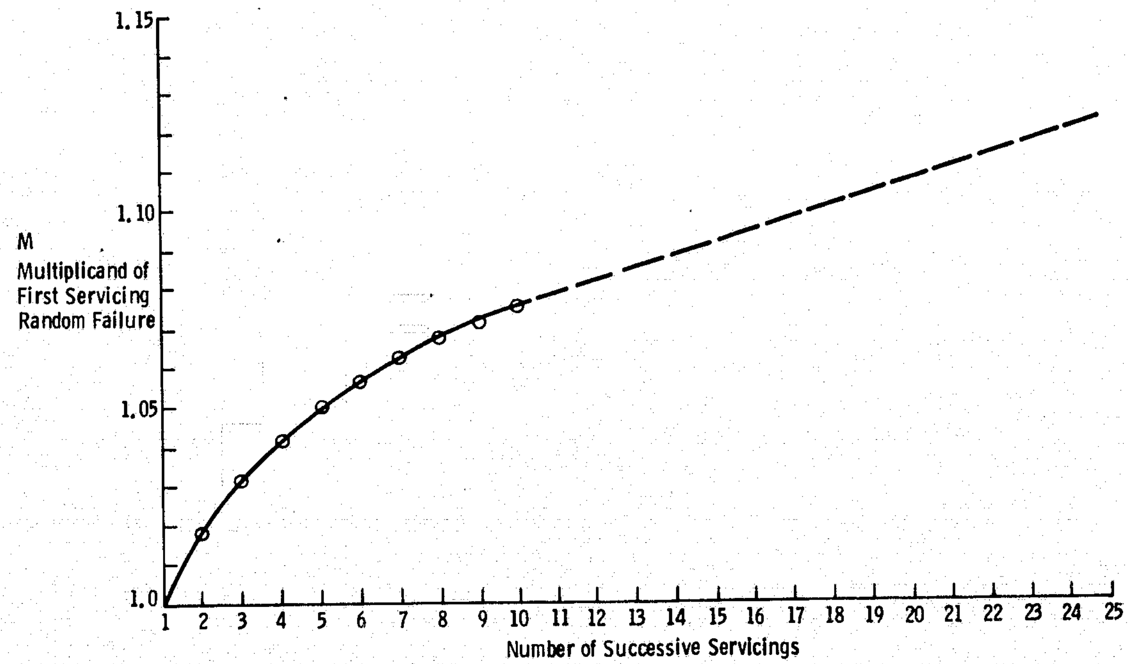

Figure IX-11 Successive Servicing Results

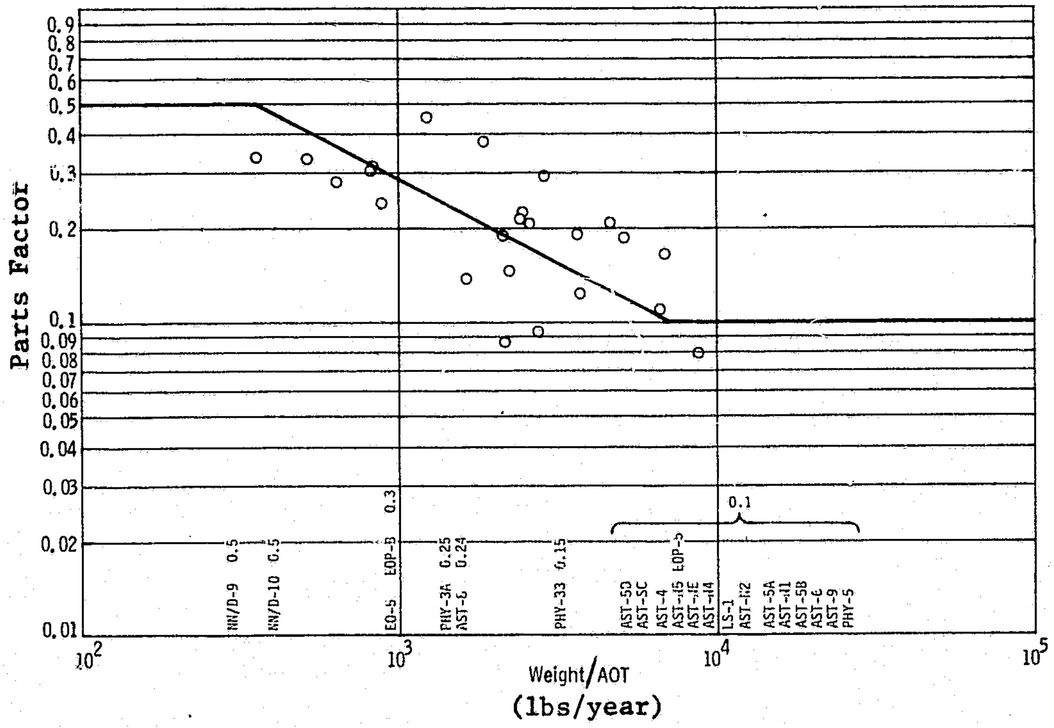

Figure IX-12 Extension of Parts Factor

For the preliminary cost analysis, the relationship

$$\text{PARTS FACTOR} = \frac{\text{Cost of Replaced Modules}}{\text{Spacecraft Unit Cost}} \approx \frac{\text{Weight of Replaced Module}}{\text{Spacecraft Total Weight}}$$

was used almost exactly to calculate part factors. However, for the baseline, final cost analysis, this relationship was modified to take into account the fact that the dollars per pound of spacecraft components vary. In particular, it was noted that the average dollars per pound for a spacecraft varied from $3000 to $7000 per pound, while the cost of monomethyl hydrazine, the primary propellant carried in all of the spacecraft considered, cost around $5 per pound, a cost difference of 3 orders of magnitude. Thus, the part factors were modified to eliminate all propellant costs.

Parts factors were also modified to rectify inconsistencies in certain spacecraft carrying modules with design lives shorter than the spacecraft design life. It was also noted that the reliability analyses used to calculate part factors included many modules with no built-in-redundancy and that actual flight modules would not be built in that manner. However, we decided to use these values since it would be too hard to attempt

to redesign the modules or reestimate the module reliability, and the values as used would tend to give higher parts factors. The effect was a too-high worst-case, estimate of maintenance costs.

A delta between on-orbit maintainable parts factors and ground-refurbishable parts factors was used to take into account additional considerations: external appendages that must be folded or removed so the satellite could fit back into the orbiter bay, additional painting, cleaning, and decontaminating, and a final acceptance test. This delta was broken down as follows:

1) Δ For appendages varied from 0.02 to 0.23 with average = 0.06
2) Δ For clean, paint, decontaminate = 0.01
3) ΔFor final qualification test = 0.04

As the baseline for the ground refurbishable handling of appendages, a parts factor was calculated assuming the removal of appendages in orbit and the replacement of the appendages on the ground. The handling of the appendage problem could have been accomplished, instead, by an increase in spacecraft DDT&E and production costs to design and install retractable appendages, but the approach selected was felt to be a less complicated and more accurate method of handling the appendage problem. The total Δ for ground refurbishment varied from 0.07 to 0.28 with an average value of 0.11

Table IX-19 presents a summary of parts factors used for both the preliminary cost analysis and the final cost analysis. Additional variations in parts factors, and their effects on the total cost picture, are discussed in the section on variations of costing parameters.

E. LOSS FACTORS

Loss factors are used to represent additional spacecraft that must be built, launched, or serviced due to potential losses such as shuttle system failures, nonreplaceable unit (NRU) failures, servicer failures, and others. Four primary loss factors have been evaluated: (1) LF = loss factor due to orbiter or tug failure to perform the mission; the same value was used for all three modes; (2) LF_1 = loss factor for retrieval in ground refurbishable missions due to failure to rendezvous and dock, or to handle appendages on spacecraft to be returned; (3) LF_2 = loss factor for number of spacecraft to be launched for ground refurbishable mode; and (4) LF_3 = Loss

Table IX-19 Parts Factor Summary

Payload No.	Payload Model Code No.	LEO Spacecraft Name	PRELIMINARY PF	PRELIMINARY pf	FINAL PF	FINAL pf
AS-03-A	AST-1B	Cosmic Background Explorer	.42	.30	.37	.25
SO-03-A	AST-3	Solar Maximum Mission	.32	.22	.27	.17
HE-09-A	AST-4	Large High Energy Observatory B	.17	.10	.16	.09
HE-03-A	AST-5A	Extended X-Ray Survey	.17	.10	.16	.09
HE-08-A	AST-5B	Large High Energy Observatory A	.17	.10	.16	.09
HE-10-A	AST-5C	Large High Energy Observatory C	.17	.10	.16	.09
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	.17	.10	.16	.09
AS-01-A	AST-6	Large Space Telescope	.17	.10	.16	.09
SO-02-A	AST-7	Large Solar Observatory	.17	.10	.16	.09
HE-11-A	AST-9A	Large High Energy Observatory D	.18	.11	.16	.09
HE-01-A	AST-9B	Large X-Ray Telescope Facility	.15	.08	.13	.06
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	.17	.10	.16	.09
AS-11-A	AST-N2	1.5m IR Telescope	.17	.10	.16	.09
AS-13-A	AST-N3	UV Survey Telescope	.17	.10	.16	.09
AS-14-A	AST-N4	1m UV - Optical Telescope	.17	.10	.16	.09
AS-17-A	AST-N5	30m IR Interferometer	.17	.10	.16	.09
HE-07-A	PHY-1A	Small High Energy Satellite	.22	.10	.21	.09
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	.27	.19	.20	.12
HE-12-A	PHY-5	Cosmic Ray Laboratory	.17	.10	.16	.09
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	.24	.11	.23	.10
EO-08-A	EO-3	Earth Observatory Satellite	.28	.17	.27	.16
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	.44	.31	.42	.29
OP-02-A	EOP-5	Gravity Gradiometer	.23	.10	.22	.09
OP-04-A	EOP-7	GRAVSAT	.50	.38	.36	.24
OP-05-A	EOP-8	Vector Magnetometer Satellite	.43	.30	.41	.28
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	.25	.15	.20	.10
		AVERAGE, LEO	.24	.15	.21	.12
		MEO/HEO Spacecraft Name				
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	.37	.24	.35	.22
AS-16-A	AST-8	Large Radio Observatory Array (2G)	.37	.24	.32	.19
AP-01-A	PHY-1B	Upper Atmosphere Explorer	.23	.12	.20	.09
AP-02-A	PHY-1C	Explorer-Medium Altitude	.24	.09	.23	.08
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	.38	.25	.34	.21
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	.28	.15	.28	.15
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	.27	.19	.23	.15
EO-12-A	EO-6	TIROS	.29	.21	.26	.18
CN-51-A	NN/D-1	INTELSAT	.49	.31	.46	.28
CN-53-A	NN/D-2B	DOMSAT B	.49	.31	.46	.28
CN-58-A	NN/D-2C	DOMSAT C	.52	.38	.45	.31
CN-54-A	NN/D-3	Disaster Warning Satellite	.56	.28	.52	.24
CN-55-A	NN/D-4	Traffic Management Satellite	.48	.33	.44	.29
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	.48	.34	.38	.24
CN-59-A	NN/D-6	Communications R&D Prototype	.72	.46	.51	.25
EO-56-A	NN/D-8	Environmental Monitoring Satellite	.28	.19	.25	.16
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	.63	.50	.50	.37
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	.64	.51	.51	.38
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	.25	.14	.24	.13
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	.32	.22	.25	.15
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	.30	.21	.23	.14
		AVERAGE, MEO & HEO	.41	.27	.35	.21
		AVERAGE	.31	.20	.28	.16

factor for on-orbit maintainable mode for failure to rendezvous and dock with a spacecraft, failure in the servicer, or failure in the NRU. Figure X-13 presents a list of the equations used to calculate loss factors.

<u>EXPENDABLE:</u>

$$LF = \ell f_{ORB} \times n \text{ FOR LEO} \Longrightarrow \ell f_{ORB} = 0.01$$

$$LF = [\ell f_{ORB} + \ell f_{TUG}] \times n \text{ FOR MEO AND HEO} \Longrightarrow \ell f_{TUG} = 0.01$$

<u>GROUND REFURBISHABLE:</u>

$$LF = LF_{EXP}$$

$$LF_1 = LF_{EXP} \text{ FOR } (n - n_f)$$

$$+ \ell f_{REND, DOCK, ACS, APPEND} \times (n - n_f) \Longrightarrow \ell f_{REND, DOCK, ACS, APPEND} = 0.02$$

$$LF_2 = MAX(LF_1, LF)$$

<u>ON-ORBIT MAINTAINABLE:</u>

$$LF = LF_{EXP}$$

$$LF_3 = LF_{1\,GR} + \ell f_{SERV} \times (n - n_f) \Longrightarrow \ell f_{SERV} = 0.03$$

$$+ \ell f_{NRU} \times (n - n_f) \Longrightarrow \ell f_{NRU} = \left.\begin{matrix} 0.01 \\ 0.02 \\ 0.03 \end{matrix}\right\} \text{DEPENDING ON SPACECRAFT}$$

Figure IX-13 Loss Factors Equations

Basically, the same loss factors were used for the preliminary and the final cost summary. However, there were several changes which did affect the loss factors. An error was found in the calculation of LF_1, due to leaving out the orbiter and tug loss factors during a retrieval mission. The number of flights over which the loss factors were applied were changed in some cases due to the changes in n and n_f, changing the total loss factors. In addition, the method in which loss factors were applied for the servicing missions were modified somewhat to allow for inability to rendezvous and dock for the servicing mission. Table IX-20 presents a summary of the loss factors used for the preliminary and final cost summaries. The main differences lie in the ground refurbishable loss factors.

Table IX-20 Loss Factor Summary

Payload No.	Payload Model Code No.	LEO Spacecraft Name	PRELIMINARY LF	PRELIMINARY LF_1	PRELIMINARY LF_2	PRELIMINARY LF_3	FINAL LF	FINAL LF_1	FINAL LF_2	FINAL LF_3
AS-03-A	AST-1B	Cosmic Background Explorer	.07	.12	.12	.36	.07	.18	.18	.42
SO-03-A	AST-3	Solar Maximum Mission	.06	.10	.10	.30	.06	.15	.15	.35
HE-09-A	AST-4	Large High Energy Observatory B	.01	.0	.01	.0	.01	.0	.01	.0
HE-03-A	AST-5A	Extended X-Ray Survey	.03	.04	.04	.16	.03	.06	.06	.18
HE-08-A	AST-5B	Large High Energy Observatory A	.03	.04	.04	.16	.03	.06	.06	.18
HE-10-A	AST-5C	Large High Energy Observatory C	.02	.02	.02	.08	.02	.03	.03	.09
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	.01	.0	.01	.0	.01	.0	.01	.0
AS-01-A	AST-6	Large Space Telescope	.12	.22	.22	.88	.12	.33	.33	.99
SO-02-A	AST-7	Large Solar Observatory	.07	.12	.12	.48	.07	.18	.18	.54
HE-11-A	AST-9A	Large High Energy Observatory D	.04	.06	.06	.24	.04	.09	.09	.27
HE-01-A	AST-9B	Large X-Ray Telescope Facility	.03	.04	.04	.16	.03	.06	.06	.18
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	.09	.16	.16	.64	.09	.24	.24	.72
AS-11-A	AST-N2	1.5m IR Telescope	.11	.20	.20	.80	.11	.30	.30	.90
AS-13-A	AST-N3	UV Survey Telescope	.06	.10	.10	.40	.06	.15	.15	.45
AS-14-A	AST-N4	1m UV - Optical Telescope	.11	.20	.20	.80	.11	.30	.30	.90
AS-17-A	AST-N5	30m IR Interferometer	.04	.06	.06	.24	.04	.09	.09	.27
HE-07-A	PHY-1A	Small High Energy Satellite	.06	.10	.10	.30	.06	.05	.06	.35
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	.02	.02	.02	.06	.02	.03	.03	.07
HE-12-A	PHY-5	Cosmic Ray Laboratory	.05	.08	.08	.32	.05	.12	.12	.36
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	.25	.48	.48	1.44	.25	.72	.72	1.68
EO-08-A	EO-3	Earth Observatory Satellite	.16	.28	.28	.84	.16	.42	.42	.98
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	.16	.28	.28	.84	.16	.42	.42	.96
OP-02-A	EOP-5	Gravity Gradiometer	.01	.0	.01	.0	.01	.0	.01	.0
OP-04-A	EOP-7	GRAVSAT	.02	.0	.02	.0	.01	0.0	.01	.0
OP-05-A	EOP-8	Vector Magnetometer Satellite	.09	.12	.12	.36	.09	.18	.18	.42
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	.18	.18	.18	.36	.09	.18	.18	.42
		MEO/HEO Spacecraft Name								
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	.06	.04	.04	.12	.06	.08	.08	.16
AS-16-A	AST-8	Large Radio Observatory Array	.08	.08	.08	.21	.08	.12	.12	.27
AP-01-A	PHY-1B	Upper Atmosphere Explorer	.04	.02	.04	.06	.04	.04	.04	.08
AP-02-A	PHY-1C	Explorer-Medium Altitude	.06	.04	.06	.12	.06	.08	.08	.16
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	.02	.0	.02	.0	.02	.0	.02	.0
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	.04	.02	.04	.06	.04	.04	.04	.08
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	.16	.12	.16	.42	.16	.24	.24	.54
EO-12-A	EO-6	TIROS	.04	.02	.04	.06	.04	.04	.04	.08
CN-51-A	NN/D-1	INTELSAT	.32	.14	.32	.49	.36	.36	.36	.81
CN-53-A	NN/D-2B	DOMSAT B	.22	.08	.22	.28	.28	.28	.28	.63
CN-58-A	NN/D-2C	DOMSAT C	.12	.06	.12	.21	.12	.12	.12	.27
CN-54-A	NN/D-3	Disaster Warning Satellite	.08	.04	.08	.14	.08	.08	.08	.18
CN-55-A	NN/D-4	Traffic Management Satellite	.28	.14	.28	.49	.28	.28	.28	.63
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	.40	.16	.40	.56	.40	.48	.48	1.08
CN-59-A	NN/D-6	Communications R&D Prototype	.06	.04	.06	.14	.06	.08	.08	.18
EO-56-A	NN/D-8	Environmental Monitoring Satellite	.14	.12	.12	.36	.14	.24	.24	.48
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	.10	.06	.10	.18	.12	.16	.16	.32
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	.14	.10	.14	.30	.16	.24	.24	.48
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	.11	.18	.18	.54	.22	.36	.36	.72
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	.08	.04	.08	.14	.20	.32	.32	.72
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	.08	.04	.08	.14	.20	.32	.32	.72

F. REPLACEMENT FACTOR

The parameter RF is the replacement factor which is used to account for extra spacecraft to replace a failed on-orbit spacecraft while it is being repaired on the ground. If a spacecraft program can afford the loss of spacecraft availability while the failed spacecraft is being repaired, then the RF value can be zero. If, on the other hand, a minimum loss of spacecraft availability is required, then RF must be at least one, and is calculated for this study by the equation:

$$RF = \left[\frac{RRT}{AOT} x n_f + 1 \right]$$

Where $[\ \ ]$ = greatest integer

AOT = average operational time

RRT = refurbishment and replacement time

n_f = on-orbit fleet size

The RRT is the period from the time that the failed spacecraft is retrieved by the orbiter and tug until it has been brought back to the ground, refurbished, scheduled for launch and replaced back in orbit.

A refurbishment time of 700 working hours was developed in the DSCSII work (see bibliography items N-12 to N-19) and can be considered satisfactory for this use.

If a replacement time of four weeks for scheduling and one week for operations is assumed, a total RRT of 0.43 years is calculated. Table IX-21 presents RF values for each of the spacecraft in the maintenance applicable set.

G. SPACECRAFT NONRECURRING AND RECURRING COSTS

Spacecraft non-recurring (DDT&E) costs - C_{NR} - and spacecraft recurring (production or unit) costs - $C_{S/C}$ - were provided by MSFC for most of the expendable spacecraft programs. These costs were then spread to the remaining expendable programs by similarity to the programs for which cost data was available. The data were received in terms of 1972 dollars and, due to one of the ground rules, were first converted to 1975 dollars before use. Factors to modify these costs were then established for both the ground refurbishable and the on-orbit maintainable modes. This was

Table IX-21 Replacement Factors

Payload No.	Payload Model Code No.	LEO Spacecraft Name	RF
AS-03-A	AST-1B	Cosmic Background Explorer	0
SO-03-A	AST-3	Solar Maximum Mission	1
HE-09-A	AST-4	Large High Energy Observatory B	0
HE-03-A	AST-5A	Extended X-Ray Survey	1
HE-08-A	AST-5B	Large High Energy Observatory A	1
HE-10-A	AST-5C	Large High Energy Observatory C	1
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	0
AS-01-A	AST-6	Large Space Telescope	1
SO-02-A	AST-7	Large Solar Observatory	1
HE-11-A	AST-9A	Large High Energy Observatory D	1
HE-01-A	AST-9B	Large X-Ray Telescope Facility	1
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	1
AS-11-A	AST-N2	1.5m IR Telescope	1
AS-13-A	AST-N3	UV Survey Telescope	1
AS-14-A	AST-N4	1m UV - Optical Telescope	1
AS-17-A	AST-N5	30m IR Interferometer	1
HE-07-A	PHY-1A	Small High Energy Satellite	1
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	0
HE-12-A	PHY-5	Cosmic Ray Laboratory	1
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	1
EO-08-A	EO-3	Earth Observatory Satellite	1
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	1
OP-02-A	EOP-5	Gravity Gradiometer	0
OP-04-A	EOP-7	GRAVSAT	0
OP-05-A	EOP-8	Vector Magnetometer Satellite	0
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	1
		MEO/HEO Spacecraft Name	
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	1
AS-16-A	AST-8	Large Radio Observatory Array	1
AP-01-A	PHY-1B	Upper Atmosphere Explorer	0
AP-02-A	PHY-1C	Explorer-Medium Altitude	0
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	0
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	1
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	1
EO-12-A	EO-6	TIROS	0
CN-51-A	NN/D-1	INTELSAT	1
CN-53-A	NN/D-2B	DOMSAT B	1
CN-58-A	NN/D-2C	DOMSAT C	1
CN-54-A	NN/D-3	Disaster Warning Satellite	1
CN-55-A	NN/D-4	Traffic Management Satellite	1
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	1
CN-59-A	NN/D-6	Communications R&D Prototype	1
EO-56-A	NN/D-8	Environmental Monitoring Satellite	1
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	1
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	1
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	1
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	1
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	1

accomplished by taking the average value of data from several previous studies (bibliography items F-5 thru F-7, K-2 thru K-8, and N-12 thru N-19) where these types of cost modifications were evaluated in detail.

The factors used to modify spacecraft unit costs for ground refurbishable (SF) and on-orbit maintainable (sf) were established for the preliminary and reexamined for the final cost analysis. The preliminary values of a 2% increase in spacecraft unit cost for ground refurbishable and a 16% increase for on-orbit maintainable were the averages for spacecraft in PUT, DSCS II and DSP studies. A review and further analysis of the data for the on-orbit maintainable spacecraft lowered this factor due to excess mission sensor redesign for one spacecraft and the removal of module/spares cost from certain spacecraft unit costs. The revised factor for on-orbit maintainable (sf) was changed from 16% to 8% and the ground refurbishable (SF) remained at 2%. It is believed that the current values are the most representative of the actual costs to be expected. Variations of these values were examined in the cost sensitivity study.

The factors used to modify spacecraft design, development and test (DDT&E) costs were also established for the preliminary analysis and then reexamined. The findings affecting unit costs also applied to the DDT&E costs. The ground refurbishable factor (D) was 2% in the preliminary analyses and remained at 2% in the final analysis. The on-orbit maintainable factor was 11% in the preliminary analyses and was revised to 4% in the final analysis.

In addition to the unit cost of the spacecraft, the recurring cost also includes the spacecraft operations. Spacecraft operations costs includes launch checkout effort for all production spacecraft and a cadre of knowledgeable personnel maintained throughout the entire mission span time for each spacecraft program. No cost was included for flight operations associated with each spacecraft as these were considered equal for all three modes.

Launch checkout (R) cost covers a recurring effort, performed at ETR/WTR, on each spacecraft and is estimated at 9 percent of the spacecraft unit cost. This percentage is the average of the actual experience from three completed spacecraft programs. Services and activities included are:

1) Vehicle receiving and inspection,
2) Prelaunch checkout,
3) Orbiter/tug mating and checkout, and
4) Prelaunch and launch countdown.

Each spacecraft program requires a cadre of knowledgeable personnel maintained from the end of production until the last launch that can be built up to the crew size necessary to perform launch operations. The cost of this sustaining effort is given by:

$$C_{KP} = C_{MY} N_P \Delta T$$

where

C_{MY} is the cost per man year

N_P is the number of personnel retained

ΔT is the number of years N_P is required

and N_P varies depending on spacecraft complexity, with the minimum being 22 people and the maximum 90. The number of years (ΔT) is the time span from first launch to last launch as shown in the mission model less the number of years it takes to produce all required spacecraft at a rate of two per year. The number of required spacecraft are equal to the number of operating cycles (n) for the expendable mode, the on-orbit fleet size (n_f) plus the replacement factor (RF) for the ground refurbishable mode, and the on-orbit fleet size (n_f) for the on-orbit maintainable mode.

$$\Delta T_{EX} = \text{Time of Last Launch} - \text{Time of First Launch} - \frac{n - n(\text{first year})}{2}$$

$$\Delta T_{GR} = \text{Time of Last Launch} - \text{Time of First Launch} - \frac{n_f + RF - n(\text{first year})}{2}$$

$$\Delta T_{OM} = \text{Time of Last Launch} - \text{Time of First Launch} - \frac{n_f - n(\text{first year})}{2}$$

H. MAINTENANCE CONCEPT COSTS

Total program cost estimates of the maintenance system concepts required to perform the on-orbit maintainable mode for the automated spacecraft were developed.

Costs of maintenance concepts were developed for four separate maintenance concepts, a pivoting arm, a general purpose manipulator, the shuttle remote manipulator system (SRMS) and extravehicular activity (EVA) from the orbiter. The first two concepts could be used both in the orbiter payload bay and on the tug, and can service all 47 programs in the maintenance applicable set. The last two can only operate from the orbiter and can service spacecraft only in LEO. In the preliminary analysis, only the pivoting arm was costed. For the final analysis, the pivoting arm cost was reevaluated and the other three methods costed. The total program cost included design, development, test and evaluation, production cost for the flight and backup articles, and operations cost for each concept. Table IX-22 presents a summary of costing considerations for each concept.

1. Maintenance Concepts - DDT&E

Design, development, test and evaluation (DDT&E) cost for each concept was estimated using parametric techniques. Parametric estimating utilizes cost estimating relationships (CERs) such as cost ratios, percentages, or a dollars per pound relationship to derive costs for each WBS element. Development cost, by WBS element, for each concept is presented in Table IX-23.

Table IX-23 On-Orbit Servicer DDT&E Costs

WBS ELEMENT	BASIS	COST ($ IN MILLIONS) PIVOTING ARM	GENERAL PURPOSE	SRMS	EVA
PROJECT MANAGEMENT	6% OF SUBTOTAL	$ 1.6	$ 1.8	$ 1.3	$ 1.0
PROJECT ENGR. & INTEGRATION	11% OF SUBTOTAL	2.7	3.0	2.1	1.7
STRUCTURES & THERMAL	SAMSO DATA	5.5	5.7	5.3	5.3
MECHANISMS	ANALOGOUS TO PDRM DATA	1.6	2.8	0.5	1.6
CONTROL ELECTRONICS	ANALOGOUS TO PDRM DATA	4.7	4.7	N/A	N/A
ASSEMBLY & CHECKOUT	5% OF HARDWARE COST	0.6	0.7	0.3	0.3
SRMS UPDATE		N/A	N/A	5.4	N/A
AIRBORNE SPARES	--	N/A	N/A	N/A	N/A
AIRBORNE SUPPORT EQUIPMENT		0.8	0.8	0.8	0.8
LOGISTICS	ANALOGOUS TO PDRM AND TUG DATA	5.6	6.7	2.8	3.8
GROUND SUPPORT EQUIPMENT	ANALOGOUS TO TUG	3.7	3.7	1.8	1.8
FACILITIES	ANALOGOUS TO TUG	0.6	0.6	0.2	0.2
OPERATIONAL SITE SERVICES	ANALOGOUS TO TUG	1.5	1.5	1.5	1.5
TOTAL		$28.9	$32.0	$22.0	$18.0

PDRM = PAYLOAD DEPLOYMENT & RETRIEVAL MECHANISM

Table IX-22 Costing Considerations

WBS ELEMENT	BASELINE-PIVOTING ARM	GENERAL PURPOSE MANIPULATOR	SRMS	EVA
PROJECT MANAGEMENT	6% OF SUBTOTAL	X	X	X
PROJECT ENGR & INTEGRATION	11% OF SUBTOTAL	X	X	X
STRUCTURES & THERMAL	400 lb STORAGE RACK	X	X	X
	50 lb RACK/TUG ADAPTER	80 lb RACK/TUG ADAPTER	N/R	N/R
MECHANISM	100 lb MANIPULATOR ARM	250 lb MANIPULATOR ARM	20 lb-END EFFECTOR	100 lb-TOOLS, RESTRAINTS, HANDHOLDS, ETC
CONTROL ELECTRONICS	30 lb 4-7.5 lb UNITS	45 lb 6-7.5 lb UNITS	N/R	N/R
ASSEMBLY & CHECKOUT	DDT&E 5% OF HARDWARE	X	X	X
	PRODUCTION 10% of HARDWARE	X	X	X
AIRBORNE SPARES	5 FLIGHT ARTICLES	X	3.5 FLIGHT ARTICLES	3.5 FLIGHT ARTICLES
	SUBSYSTEMS-PARTIAL	X	0	0
AIRBORNE SUPPORT EQUIPMENT	200 lb CRADLE-RACK	X	X	X
	250 lb SERVICING PLATFORM°	X	X	X
LOGISTICS	LOGISTICS MANAGEMENT	X	X	X
	INVENTORY CONTROL	X	X	X
	O&M MANUALS	X	0	0
	TRAINERS (1 ETR, 1 WTR)	X	0	0
	TRAINING	X	X	ⓧ
GROUND SUPPORT EQUIPMENT	MECHANICAL-49 UNITS	X	X	X
	ELECTRICAL-15 UNITS	X	ELECTRICAL-3 UNITS	ELECTRICAL-3 UNITS
FACILITIES	REARRANGEMENT	X	0	0
OPERATIONAL SITE SERVICES	LAUNCH OPERATIONS	X	0	0
	FLIGHT OPERATIONS	X	X	ⓧ
	MAINTENANCE	X	0	0

LEGEND:

- X REQUIREMENTS IDENTICAL TO BASELINE
- 0 REQUIREMENTS REDUCED FROM BASELINE
- ⓧ REQUIREMENTS INCREASED FROM BASELINE
- ° EQUIPMENT WAS LATER DELETED FROM THE PIVOTING ARM. HOWEVER, COSTS WERE NOT CHANGED.

Project management is estimated at six percent of all other WBS elements, cost and project engineering and integration at 11 percent of all WBS elements excluding project management. These percentages have been developed from previous contract history.

Structures and thermal WBS elements consist of a stowage rack for all four maintenance concepts and a rack/tug adapter for the pivoting arm and general purpose manipulator concepts. The SAMSO cost model for structure was used to cost this element.

Mechanisms WBS element consists of a manipulator arm for the pivoting arm and general purpose manipulator concepts, a special end effector for the SRMS concept and various tools, restraints, hand holds, etc, for the EVA concept. MMC cost data for a previous study on payload deployment and retrieval mechanism (now called the shuttle remote manipulator system) was adjusted to criteria in the areas of technical requirements, number of development units, length of development program, fiscal year, etc. This adjusted development cost was the data point to derive a straight line log-log plot of development cost versus mechanisms weight that encompasses all four maintenance concepts.

Control electronics WBS element consists of servo-electronics subassemblies (one at each joint) and the required software for the pivoting arm and general purpose manipulator concepts. The electronic subassemblies are similar to those in the payload deployment and retrieval mechanism (PDRM). This PDRM cost data was adjusted as discussed under mechanisms to derive DDT&E costs. Software development was estimated based on similarity to other programs of comparable scope.

SRMS update WBS element consists of the additional development cost to the current shuttle remote manipulator system program to acquire the precision and fidelity (add seventh degree of freedom, increased precision of joints, added mechanical and electrical components, etc) required to perform module exchange.

Assembly and checkout WBS element consists of the design and fabrication of assembly tools and the development of acceptance test procedures. This element is estimated at five percent of all subsystems hardware cost.

Airborne support equipment WBS element consists of two items: a cradle to mount the module stowage rack in the orbiter and a spacecraft servicing

platform with rotation capability that would support the spacecraft during module exchange. Airborne support equipment development was estimated using the mechanisms data with reductions for complexity. Differences with airborne support equipment among the various concepts is discussed in more detail in chapter IV. The difference in the items has led to difference in the launch cost reimbursement policy effects on the various concepts.

Logistics WBS element consists of logistics management, inventory control, operations and maintenance manuals, trainers, and training. All these functions, except trainers, were estimated from Space Tug Systems Study-Storable data as a reference point and adjusted based on relative complexity. Two trainers, one each at ETR and WTR, provided for all four maintenance concepts with the EVA concept requiring a third unit for neutral buoyancy training. Trainers are estimated as the equivalent of a flight unit and then adjusted for fidelity requirements.

Ground support equipment WBS element consists of the development of GSE and the production of GSE required for ETR and WTR. A list of GSE and quantities of each item for the pivoting arm concept is shown in Table IX-24. GSE development was estimated based on similarity to space tug GSE items.

Table IX-24 Ground Support Equipment

ITEM	QUANTITY
ACCESS EQUIPMENT	3
PORTABLE HOIST/CRANE	3
SLING SETS-SERVICER	4
SLING SETS-ADAPTER	4
SLING SETS-CRADLE	4
ADAPTER INSTALLATION TOOL KIT	3
PORTABLE TRANSPORTER	4
CRADLE AND RESTRAINTS	4
ELECTRICAL TEST SET	3
ORDNANCE TEST SET	3
LATCH MECHANISM TEST SET	3
TUG/SERVICER INTERFACE SIMULATOR	3
COMMUNICATIONS TEST SET	3
PROTECTIVE COVERS	5
ALIGNMENT SET	3
STORAGE SUPPORT FIXTURE	10

Facilities WBS element consists of rearrangement costs to existing facilities (contractor and government) and assumes no new facilities. This cost is for rearrangement of 3000 sq ft at ETR and WTR plus modification and

rearrangement of contractor manufacturing and test facilities. The estimates were derived from historical experience on programs of a similar nature.

Operational site services WBS element consists of mission planning and flight support software development. These items were costed based on relative complexity and number of operations as compared with space tug effort.

2. Maintenance Concepts - Production

Unit and production costs for the four maintenance concepts are presented in Table IX-25.

Table IX-25 On-Orbit Servicer Unit and Production Costs

WBS ELEMENT	COST ($ IN M)							
	PIVOTING ARM		GENERAL PURPOSE		SRMS		EVA	
	UNIT	PRODUCTION	UNIT	PRODUCTION	UNIT	PRODUCTION	UNIT	PRODUCTION
PROJECT MANAGEMENT	0.10	0.97	0.13	1.26	0.07	0.59	0.08	0.63
PROJECT ENGR. & INTEGRATION	0.17	1.60	0.21	2.08	0.12	0.98	0.14	1.05
STRUCTURES & THERMAL	0.98	2.94	1.04	3.12	0.90	2.70	0.90	2.70
MECHANISMS	0.24	0.72	0.45	1.35	0.08	0.24	0.24	0.72
CONTROL ELECTRONICS	0.17	0.51	0.25	0.75	N/A	N/A	N/A	N/A
ASSEMBLY & CHECKOUT	0.14	0.42	0.17	0.51	0.10	0.30	0.11	0.33
SRMS UPDATE	N/A	N/A	N/A	N/A	N/A	9.40	N/A	N/A
AIRBORNE SPARES		8.45		11.72		4.23		4.30
AIRBORNE SUPPORT EQUIPMENT		1.46		1.46		1.46		1.46
LOGISTICS		N/A		N/A		N/A		N/A
GROUND SUPPORT EQUIPMENT		N/A		N/A		N/A		N/A
FACILITIES		N/A		N/A		N/A		N/A
OPERATIONAL SITE SERVICES		N/A		N/A		N/A		N/A
TOTAL	1.80	17.07	2.25	22.25	1.27	10.50	1.47	11.19

Project management is estimated at six percent of all other WBS elements, cost and project engineering and integration at 11 percent of all WBS elements excluding project management. These percentages have been developed from previous contract history.

Unit subsystem WBS elements (structures, mechanisms, and control electronics) costs were estimated using the same data sources and techniques as previously described for these elements under DDT&E. The production costs for all concepts are all based on three units with no application of an improvement curve.

SRMS update WBS element consists of the delta cost to the existing flight units, caused by the added requirements previously described under DDT&E and the addition of one flight unit because of the increased number of operations due to performing module exchange.

Assembly and checkout WBS element consists of assembly, installation, checkout and acceptance testing. This effort is estimated at ten percent of the subsystem hardware cost.

Airborne spares WBS element consists of complete flight article spares and subsystem component spares. For the pivoting arm and general purpose manipulator concepts, airborne spares are five units consisting of one spare unit and four units calculated as reliability losses. For SRMS and EVA, which are only used for low earth orbit missions, the airborne spares are 3.5 units consisting of 1 spare unit and 2.5 units calculated as reliability losses. Subsystem component spares for all concepts are calculated at 20 percent of mechanisms flight hardware costs and 30 percent of control electronics flight hardware costs.

Airborne support equipment WBS element costs were estimated using the same data sources and techniques as previously described for this element under DDT&E. The production cost is based on 5.6 units with no application of improvement curve.

3. Maintenance Concepts-Operations

Operations category summarizes the cost of launch operations, flight operations, maintenance (scheduled and unscheduled), refurbishment, management and supporting functions. Table IX-26 presents operations costs, by WBS element,

Table IX-26 On-Orbit Servicer Operations Costs

WBS ELEMENT	BASIS	COST ($ IN MILLIONS) PIVOTING ARM	GENERAL PURPOSE	SRMS	EVA
PROJECT MANAGEMENT	6% OF SUBTOTAL	$ 3.2	$ 3.6	$ 2.3	$ 2.4
PROJECT ENGR. & INTEGRATION	11% OF SUBTOTAL	5.3	6.0	3.7	3.9
STRUCTURES & THERMAL	SAMSO DATA	N/A	N/A	N/A	N/A
MECHANISMS	ANALOGOUS TO PDRM DATA	N/A	N/A	N/A	N/A
CONTROL ELECTRONICS	ANALOGOUS TO PDRM DATA	N/A	N/A	N/A	N/A
ASSEMBLY & CHECKOUT	% OF HARDWARE COST	N/A	N/A	N/A	N/A
SRMS UPDATE	--	N/A	N/A	N/A	N/A
AIRBORNE SPARES	--	N/A	N/A	N/A	N/A
AIRBORNE SUPPORT EQUIPMENT	--	N/A	N/A	N/A	N/A
LOGISTICS	ANALOGOUS TO PDRM AND TUG DATA	4.9	4.9	4.9	6.9
GROUND SUPPORT EQUIPMENT	ANALOGOUS TO TUG	0.5	0.5	0.2	0.2
FACILITIES	ANALOGOUS TO TUG	N/A	N/A	N/A	N/A
OPERATIONAL SITE SERVICES	ANALOGOUS TO TUG	42.6	48.9	28.9	37.4
TOTAL		$56.5	$63.9	$40.0	$50.8

PDRM = PAYLOAD DEPLOYMENT & RETRIEVAL MECHANISM

for the four maintenance concepts. These costs cover 12 years at ETR and nine years at WTR.

Project management is estimated at six percent of all other WBS elements, cost and project engineering and integration at 11 percent of all WBS elements excluding project management. These percentages have been developed from previous contract history.

Logistics WBS element consists of implementing the various logistics management functions and inventory control techniques developed during the DDT&E phase. Also included is effort to update operations and maintenance manuals to incorporate information gathered through the operations phase. The nature of training makes it a continuing task which extends through all of the operations phase. This is due to attrition of personnel as well as the variations in payloads as a result of the mission model. These tasks were estimated as to relative complexity with comparable items in the Space Tug Systems Study. The EVA concept cost includes effort to simulate each of the maintenance missions in a neutral buoyancy tank. This task is estimated at two million dollars for the 12 years of operations.

Operational site services WBS element consists of the repetitive tasks and functions of launch operations, flight operations and maintenance/refurbishment. Launch operations includes servicer receiving and inspection, prelaunch checkout, orbiter/tug mating and checkout and launch countdown. Flight operations includes mission planning, flight control, data reduction, analysis and documentation. Maintenance/refurbishment is restoring the reusable servicer, after each mission, to a flight readiness condition for subsequent missions. Cost estimates for the functions just described are based on the manpower (engineering, technical and support) and material required to sustain the mission model flight schedule. Space Tug Systems Study operations costs were examined in detail and inapplicable costs such as propellants and gases were deleted from the base data. Analyses of relative complexity and number of operations of each servicer concept as compared to the tug established ratios to apply against the basic data. Cost estimates for operational site services showing per year costs by launch site and totals for the 12-year operational phase for each maintenance concept is given in Tables IX-27, IX-28, IX-29, and IX-30. EVA operational site services, Table IX-30, includes the crew EVA operations element at $60K per service mission (see chapter XI, item B-1).

Table IX-27 Operational Site Services, Pivoting Arm

WBS ELEMENT	($ IN MILLIONS)		
	ETR/Yr	WTR/Yr	TOTAL
SITE SERVICES AND SUPPORT	$0.30	$0.30	$ 6.30
MISSION PLANNING	0.50	0.30	8.70
FLIGHT CONTROL	0.50	--	6.00
FLIGHT EVALUATION	0.10	0.10	2.10
SCHEDULED MAINTENANCE	0.20	0.10	3.30
UNSCHEDULED MAINTENANCE	0.30	0.10	4.50
POSTFLIGHT CHECKOUT	0.10	0.10	2.10
TUG MATING AND CHECKOUT	0.30	0.20	5.40
DEPOT MAINTENANCE	0.20	0.20	4.20
TOTAL	$2.50	$1.40	$42.60

Table IX-28 Operational Site Services, General Purpose Manipulator

WBS ELEMENT	($ IN MILLIONS)		
	ETR/Yr	WTR/Yr	TOTAL
SITE SERVICES AND SUPPORT	$0.30	$0.30	$ 6.30
MISSION PLANNING	0.50	0.30	8.70
FLIGHT CONTROL	0.50	--	6.00
FLIGHT EVALUATION	0.10	0.10	2.10
SCHEDULED MAINTENANCE	0.20	0.10	3.30
UNSCHEDULED MAINTENANCE	0.50	0.20	10.80
POSTFLIGHT CHECKOUT	0.10	0.10	2.10
TUG MATING AND CHECKOUT	0.30	0.20	5.40
DEPOT MAINTENANCE	0.20	0.20	4.20
TOTAL	$2.70	$1.50	$48.90

Table IX-29 Operational Site Services, SRMS

WBS ELEMENT	($ IN MILLIONS)		
	ETR/Yr	WTR/Yr	TOTAL
SITE SERVICES AND SUPPORT	$0.10	$0.10	$ 2.10
MISSION PLANNING	0.50	0.30	8.70
FLIGHT CONTROL	0.50	--	6.00
FLIGHT EVALUATION	0.10	0.10	2.10
SCHEDULED MAINTENANCE	0.20	0.10	3.30
UNSCHEDULED MAINTENANCE	0.20	0.10	3.30
POSTFLIGHT CHECKOUT	0.06	0.06	1.40
ORBITER MATING AND CHECKOUT	0.03	0.03	0.60
DEPOT MAINTENANCE	0.07	0.07	1.40
TOTAL	$1.76	$0.86	$28.90

Table IX-30 Operational Site Services, EVA

WBS ELEMENT	($ IN MILLIONS)		
	ETR/Yr	WTR/Yr	TOTAL
SITE SERVICES AND SUPPORT	$0.10	$0.10	$ 2.10
MISSION PLANNING	0.70	0.40	12.00
FLIGHT CONTROL	0.50	--	6.00
FLIGHT EVALUATION	0.10	0.10	2.10
SCHEDULED MAINTENANCE	0.10	0.10	2.10
UNSCHEDULED MAINTENANCE	0.10	0.10	2.10
POSTFLIGHT CHECKOUT	0.03	0.03	0.70
CREW EVA OPERATIONS	0.70	0.20	8.90
DEPOT MAINTENANCE	0.07	0.07	1.40
TOTAL	$2.40	$1.10	$37.40

Results of the costing of all the maintenance concepts are summarized in Table IX-31.

Table IX-31 Maintenance Concept Cost Summary

MAINTENANCE CONCEPT	COST (DOLLARS IN MILLIONS)			TOTAL
	DDT&E	PRODUCTION	OPERATIONS	
PIVOTING ARM	29	17	57	103
GENERAL PURPOSE MANIPULATOR	32	22	64	118
SHUTTLE REMOTE MANIPULATOR SYSTEM	22	20	40	82
EVA	18	11	51	80

I. FINAL MAINTENANCE MODE COST SUMMARY

Results of the final cost analysis can be presented in a myriad of formats. Figures IX-14, IX-15 and IX-16 present total program costs for all 47 programs in the maintenance applicable set if all programs are flown in the expendable, ground refurbishable, and on-orbit maintainable modes. Figure IX-14 presents a summary for LEO spacecraft only, Figure IX-15 presents the summary for MEO and HEO spacecraft only, and Figure IX-16 presents the summary for all of the spacecraft. In these summaries, the maintenance concept costs have been excluded. (This same type of data has been calculated for each of the 47 individual programs in the maintenance applicable set and is summarized in Table IX-32.) It can be seen from these figures that a significant savings can be realized ($\approx$9 billion dollars) if all spacecraft are flown in an on-orbit maintainable mode. However, it was noted during the preliminary cost analysis that the most economical method of flying this mission model was to provide the capability to fly all three of these modes, and to fly each particular program in the mode that is the least expensive for it. One of the results of this study showed that if a particular program has only one mission, it will be cheaper to fly expendably. All other missions were less expensive to fly in an on-orbit maintainable mode. None of the programs were less expensive in a ground refurbishable mode; however, this could change depending on the final form of the LCRP used by NASA. Table IX-33 presents a summary of LEO spacecraft expendable mode program

SPACECRAFT PROGRAM			EXPENDABLE	GROUND REFURBISHABLE	ON-ORBIT MAINTAINABLE
ORBITER			1.0	1.6	0.4
TUG			—	—	—
SPACECRAFT			15.3	8.1	7.4
	DDT&E		4.4	4.5	4.6
	PRODUCTION		9.7	3.4	2.6
	OPERATIONS		1.2	0.2	0.2
		LAUNCH C/O	0.9		
		SUSTAIN	0.3		
SPACECRAFT/MODULE REFURBISHMENT	OPERATIONS		N/A	2.8	1.6
		REFURB		1.6	
		REPLACE			0.9
		LAUNCH C/O		0.7	0.1
		SUSTAIN		0.5	0.6
TOTAL			16.3	12.5	9.4
ZERO LAUNCH COSTS (ZLC)			15.3	10.9	9.0
				(Δ = 4.4 ZLC)	(Δ = 6.3 ZLC)
				(Δ = 3.8)	(Δ = 6.9

Figure IX-14 Cost Summary, LEO Spacecraft

SPACECRAFT PROGRAM		EXPENDABLE	GROUND REFURBISHABLE	ON-ORBIT MAINTAINABLE
ORBITER		1.0	1.4	0.8
TUG		0.1	0.2	0.1
SPACECRAFT	DDT&E	7.4: 2.5	4.6: 2.5	4.5: 2.5
	PRODUCTION	4.3	2.0	1.8
	OPERATIONS	0.6 (LAUNCH C/O 0.4; SUSTAIN 0.2)	0.1	0.2
SPACECRAFT/MODULE REFURBISHMENT	OPERATIONS	N/A	1.5: REFURB 0.9; LAUNCH C/O 0.3; SUSTAIN 0.3	1.0: REPLACE 0.6; LAUNCH C/O 0.1; SUSTAIN 0.3
TOTAL		8.5	7.7	6.4
ZERO LAUNCH COST (ZLC)		7.4	6.1	5.5
			(Δ= 1.3 ZLC) (Δ = 0. 8)	(Δ = 1.9 ZLC) (Δ = 2. 1)

Figure IX-15 Cost Summary, MEO and HEO Spacecraft

SPACECRAFT PROGRAM		EXPENDABLE		GROUND REFURBISHABLE		ON-ORBIT MAINTAINABLE	
ORBITER			2.0		3.0		1.2
TUG			0.1		0.2		0.1
SPACECRAFT	DDT&E	22.7	6.9	12.7	7.0	12.0	7.2
	PRODUCTION		14.0		5.4		4.4
	OPERATIONS		1.8 (LAUNCH C/O 1.3; SUSTAIN 0.5)		0.3		0.4
SPACECRAFT/MODULE REFURBISHMENT	OPERATIONS		N/A	4.3	REFURB 2.5	2.5	REPLACE 1.5
					LAUNCH C/O 1.0		LAUNCH C/O 0.1
					SUSTAIN 0.8		SUSTAIN 0.9
TOTAL			24.8		20.2		15.8
ZERO LAUNCH COSTS (ZLC)			22.7		17.0		14.5
					(Δ = 5.7 ZLC) (Δ = 4.6)		(Δ = 8.2 ZLC) (Δ = 9.0)

Figure IX-16 Cost Summary, All Spacecraft

Table IX-32 Individual Spacecraft Program Cost Summary

Payload No.	Payload Model Code No.	LEO Spacecraft Name	TOTAL PROGRAM COST (C&M) EXP	GR	OOM
AS-03-A	AST-1B	Cosmic Background Explorer	145.3	133.6	105.8
SO-03-A	AST-3	Solar Maximum Mission	491.6	408.5	324.0
HE-09-A	AST-4	Large High Energy Observatory B	242.4	247.2	253.9
HE-03-A	AST-5A	Extended X-Ray Survey	515.7	482.4	381.3
HE-08-A	AST-5B	Large High Energy Observatory A	544.9	509.3	403.2
HE-10-A	AST-5C	Large High Energy Observatory C	313.6	334.9	268.7
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	288.9	294.6	302.6
AS-01-A	AST-6	Large Space Telescope	1659.1	1005.6	683.0
SO-02-A	AST-7	Large Solar Observatory	1079.9	780.7	555.3
HE-11-A	AST-9A	Large High Energy Observatory D	634.8	550.2	427.6
HE-01-A	AST-9B	Large X-Ray Telescope Facility	614.1	575.2	442.6
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	1058.1	698.0	492.4
AS-11-A	AST-N2	1.5m IR Telescope	978.4	609.9	428.1
AS-13-A	AST-N3	UV Survey Telescope	504.4	391.1	290.1
AS-14-A	AST-N4	1m UV - Optical Telescope	795.1	511.6	349.7
AS-17-A	AST-N5	30m IR Interferometer	388.5	349.1	246.8
HE-07-A	PHY-1A	Small High Energy Satellite	619.5	421.4	289.2
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	185.5	172.7	168.8
HE-12-A	PHY-5	Cosmic Ray Laboratory	1033.4	772.1	581.0
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	873.6	642.2	345.1
EO-08-A	EO-3	Earth Observatory Satellite	1982.0	1338.9	929.4
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	196.4	182.9	135.8
OP-02-A	EOP-5	Gravity Gradiometer	220.2	224.5	230.7
OP-04-A	EOP-7	GRAVSAT	251.6	252.8	260.6
OP-05-A	EOP-8	Vector Magnetometer Satellite	197.5	178.4	158.3
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	529.1	427.1	342.1
		TOTAL, LEO	16343.6	12494.9	9396.1
		MEO/HEO Spacecraft Name			
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	121.3	136.0	111.8
AS-16-A	AST-8	Large Radio Observatory Array	253.9	250.3	210.4
AP-01-A	PHY-1B	Upper Atmosphere Explorer	126.1	120.8	119.4
AP-02-A	PHY-1C	Explorer-Medium Altitude	241.7	225.6	221.1
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	260.8	266.0	272.1
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	303.5	326.6	277.0
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	537.0	462.2	345.7
EO-12-A	EO-6	TIROS	169.5	160.7	158.6
CN-51-A	NN/D-1	INTELSAT	795.6	748.2	635.6
CN-53-A	NN/D-2B	DOMSAT B	697.4	676.1	567.0
CN-58-A	NN/D-2C	DOMSAT C	204.1	214.4	184.4
CN-54-A	NN/D-3	Disaster Warning Satellite	204.7	220.7	184.1
CN-55-A	NN/D-4	Traffic Management Satellite	289.1	291.2	252.8
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	433.6	413.1	372.3
CN-59-A	NN/D-6	Communications R&D Prototype	284.8	306.6	246.1
EO-56-A	NN/D-8	Environmental Monitoring Satellite	392.9	338.7	285.4
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	199.8	206.8	179.8
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	233.6	236.8	199.5
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	1502.0	1060.1	863.6
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	632.0	533.4	375.5
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	642.1	532.3	376.0
		TOTAL, MEO/HEO	8525.1	7726.6	6438.2
		TOTAL	24869.1	20221.5	15834.3

costs, savings of on-orbit maintainable mode over expendable mode, and percent savings. Each is ranked from highest to lowest. Table IX-34 presents the same data for the MEO and HEO spacecraft.

Tables IX-33 and IX-34 show that by proper mode selection for each of the spacecraft programs, a savings of 7 billion dollars in LEO programs and a savings of 2.1 billion dollars on MEO and HEO programs (for a total of 9.1 billion dollars) can be realized over flying all of these programs in an expendable mode. Should NASA not develop the capability to perform any on-orbit maintenance, a savings of 3.9 billion dollars for LEO spacecraft and a savings of 0.9 billion dollars for MEO and HEO spacecraft, a total of 4.8 billion dollars, could still be realized by developing the capability to perform ground refurbishment.

The savings for LEO and for MEO and HEO spacecraft have been presented separately since not all of the four maintenance concepts costed can service all the spacecraft. Only the pivoting arm and general purpose manipulator have the capability of being flown in the orbiter or attached to the tug. (They could also be attached to some sort of free flying teleoperator system.) The SRMS and EVA could only be performed near the orbiter. It may be possible to return some MEO spacecraft to the orbiter, service them with the SRMS or EVA, and return them to their orbit, all utilizing one tug; however, it is suggested that additional work be performed on this mode of operation. Approximately 0.8 billion of the 2.1 billion dollars savings for MEO and HEO spacecraft are in this class.

It has been suggested by several sources that should the savings gained by going from an expendable mode spacecraft to an on-orbit maintainable mode spacecraft not prove sufficiently high, then a spacecraft program might still elect to fly in the proven mode of expendable. The last column in Tables IX-33 and IX-34 show the percent savings (of expendable cost) to be realized by utilizing the on-orbit maintainable mode. Spacecraft programs are ranked from highest to lowest. Thus for LEO spacecraft, should only those spacecraft which demonstrate a minimum 30 percent savings over expendable be selected to be flown in the on-orbit maintainable mode, the total savings would be about 6.4 billion instead of the 7.0 billion dollars. For a minimum 20 percent savings, the total savings would be 6.9 billion dollars. For MEO and HEO spacecraft, a 30 percent minimum would mean about 1.3 billion

Table IX-33 LEO Spacecraft Summary

EXP COST		Σ	SAVINGS		Σ	% SAVINGS		Σ	
EO-3	$ 1,982.0	$ 1,982.0	EO-3	$1052.6	$1052.6	LS-1	60.5 %		
AST-6	1,659.1	3,641.1	AST-6	976.1	2028.7	AST-6	58.8		
AST-7	1,079.9	4,721.0	AST-N1	565.7	2594.4	AST-N2	56.2		
AST-N1	1,058.1	5,779.1	AST-N2	550.3	3144.7	AST-N4	56.0		
PHY-5	1,033.4	6,812.5	LS-1	528.5	3673.2	AST-N1	53.5		
AST-N2	978.4	7,790.9	AST-7	524.6	4197.8	PHY-1A	53.3		
LS-1	873.6	8,664.5	PHY-5	452.4	4650.2	EO-3	53.1		
AST-N4	795.1	9,459.6	AST-N4	445.4	5095.6	AST-7	48.6		
AST-9A	634.8	10,094.4	PHY-1A	330.3	5425.9	PHY-5	43.8		
PHY-1A	619.5	10,713.9	AST-N3	214.3	5640.2	AST-N3	42.5		
AST-9B	614.1	11,328.0	AST-9A	207.2	5847.4	AST-N5	36.5		
AST-5B	544.9	11,872.9	NN/D-14	187.0	6034.4	NN/D-14	35.3		
NN/D-14	529.1	12,402.0	AST-9B	171.5	6205.9	AST-3	34.1		
AST-5A	515.7	12,917.7	AST-3	167.6	6373.5	AST-9A	32.6		
AST-N3	504.4	13,422.1	AST-5B	141.7	6515.2	EO-5	30.9	$6371.1	30%
AST-3	491.6	13,913.7	AST-N5	141.7	6656.9	AST-9B	27.9		
AST-N5	388.5	14,302.2	AST-5A	134.4	6791.3	AST-1B	27.2		
AST-5C	313.6	14,615.8	EO-5	60.6	6851.9	AST-5A	26.1		
AST-5D	288.9	14,904.7	AST-5C	44.9	6896.8	AST-5B	26.0	6891.4	25%
EOP-7	251.6	15,156.3	AST-1B	39.5	6936.3	EOP-8	19.8	6930.6	15%
AST-4	242.4	15,398.7	EOP-8	39.2	6975.5	AST-5C	14.3	6975.5	10%
EOP-5	220.2	15,618.9	PHY-2A	16.7	6992.2	PHY-2A	9.0		
EOP-8	197.5	15,816.4	AST-4	0		AST-4	0		
EO-5	196.4	16,012.8	AST-5D	0		AST-5D	0		
PHY-2A	185.5	16,198.3	EOP-5	0		EOP-5	0		
AST-1B	145.3	16,343.6	EOP-7	0		EOP-7	0		
TOTAL	$16,343.6		TOTAL		$6992.2				
ALL COST AND SAVINGS ARE MILLIONS OF DOLLARS.									

Table IX-34 MEO and HEO Spacecraft Summary

EXP COST		Σ	SAVINGS		Σ	% SAVINGS		Σ	
NN/D-11	$1502.0	$1502.0	NN/D-11	$638.4	$ 638.4	NN/D-11	42.5 %		
NN/D-1	795.6	2297.6	NN/D-13	266.1	904.5	NN/D-13	41.4		
NN/D-2B	697.4	2995.0	NN/D-12	256.5	1161.0	NN/D-12	40.6		
NN/D-13	642.1	3637.1	EO-4	191.3	1352.3	EO-4	35.6	$1268.5	30%
NN/D-12	632.0	4269.1	NN/D-1	160.0	1512.3	NN/D-8	27.4	1459.8	25%
EO-4	537.0	4806.1	NN/D-2B	130.4	1642.7	NN/D-1	20.1	1619.8	20%
NN/D-5	433.6	5239.7	NN/D-8	107.5	1750.2	NN/D-2B	18.7		
NN/D-8	392.9	5632.6	NN/D-5	61.3	1811.5	AST-8	17.1	1793.7	15%
PHY-3B	303.5	5936.1	AST-8	43.5	1855.0	NN/D-10	14.6		
NN/D-4	289.1	6225.2	NN/D-6	38.7	1893.7	NN/D-5	14.1		
NN/D-6	284.8	6510.0	NN/D-4	36.3	1930.0	NN/D-6	13.6		
PHY-3A	260.8	6770.8	NN/D-10	34.1	1964.1	NN/D-4	12.6		
AST-8	253.9	7024.7	PHY-3B	26.5	1990.6	NN/D-3	10.1		
PHY-1C	241.7	7266.4	PHY-1C	20.6	2011.2	NN/D-9	10.0	2004.7	10%
NN/D-10	233.6	7500.0	NN/D-3	20.6	2031.8	NN/D-2C	9.7		
NN/D-3	204.7	7704.7	NN/D-9	20.0	2051.8	PHY-3B	8.7		
NN/D-2C	204.1	7908.8	NN/D-2C	19.7	2071.5	PHY-1C	8.5		
NN/D-9	199.8	8108.6	EO-6	10.9	2082.4	AST-1C	7.8		
EO-6	169.5	8278.1	AST-1C	9.5	2091.9	EO-6	6.4		
PHY-1B	126.1	8404.2	PHY-1B	6.7	2098.6	PHY-1B	5.3		
AST-1C	121.3	8525.3	PHY-3A	0	2098.6	PHY-3A	0		
ALL COST AND SAVINGS ARE MILLIONS OF DOLLARS.									

instead of the 2.1 billion dollars savings. A 20 percent minimum could mean a 1.6 billion dollar savings, instead of the full 2.1 billion dollars.

J. ADDITIONAL COSTING CONSIDERATIONS

1. Design Failure

In the calculation of the parts factors, two types of failures were considered; the random failures and the wear-out failures. Spacecraft have, however, through the years also demonstrated that there is a third type of failure, the design failure. Design failures consist of those failures that occur due to an error or oversight in design, manufacture or testing that could have been corrected, if noticed, prior to launch. If a design failure occurs in a module or component of one spacecraft, the same design failure will usually be present in the same modules, or components, in all identical spacecraft. While analytical methods have been developed to treat random and wear-out failures, none have been developed for design failures. To investigate the effects of design failure, it was decided to investigate past data on spacecraft programs and to determine some method of applying this historical data to the maintenance applicable set of spacecraft. It was suggested to us by the COMSAT Corporation, and we agreed to use the assumption, that two design failures would appear and could be corrected by the third year of each mission, and that one would be capable of a work around not requiring either a service mission or another expendable mission. In order to obtain some estimate of how design failures might affect program cost, it was decided to either replace all spacecraft in orbit for each program three years after the first launch, which had not been replaced by a normal failure (expendable mode) or to service all spacecraft in orbit for each program three years after the initial launch which had not yet been serviced for a normal failure (on-orbit maintainable mode). A review of the normal mission model showed that this affected eight programs, all in geostationary orbits. All other programs had expected servicings before or at the three year time period and would not require additional missions. Table IX-35 presents a new summary for the MEO and HEO spacecraft, listing the expendable costs, savings for on-orbit maintenance, and percent savings. It shows that the savings in MEO and HEO could increase from 2.1 billion up to 2.5 billion dollars during this era if these projected design failures are taken into account. While this type of analysis only provides a very rough estimate, at

Table IX-35 Design Failures Effects for MEO and HEO Spacecraft

EXP COST		Σ	SAVINGS		Σ	% SAVINGS		Σ	
NN/D-11	$1502.0	$1502.0	NN/D-11	$638.4	$ 638.4	NN/D-11	42.5 %		
NN/D-1*	1056.2	2558.2	NN/D-1*	299.4	937.8	NN/D-13	41.4		
NN/D-2B*	814.5	3372.7	NN/D-13	266.1	1203.9	NN/D-12	40.6	$1161	40%
NN/D-13	642.1	4014.8	NN/D-12	256.5	1460.4	EO-4	35.6	1352.3	30%
NN/D-12	632.0	4646.8	NN/D-2B*	196.9	1656.9	NN/D-1*	28.3		
EO-4	537.0	5183.8	EO-4	191.3	1848.2	NN/D-8	27.4	1759.2	25%
NN/D-5*	523.6	5707.4	NN/D-5*	109.1	1957.2	NN/D-2B*	24.1		
NN/D-8	392.9	6100.3	NN/D-8	107.5	2064.8	NN/D-5*	20.8		
NN/D-4*	343.4	6443.7	NN/D-6*	65.7	2130.5	NN/D-6*	20.3	2130.5	20%
NN/D-6*	324.4	6768.1	NN/D-4*	59.7	2190.2	NN/D-3*	19.3		
PHY-3B	303.5	7071.6	NN/D-3*	48.8	2239.0	NN/D-2C*	18.5		
NN/D-2C*	262.8	7334.4	NN/D-2C*	48.7	2287.7	NN/D-9*	18.3		
PHY-3A	260.8	7595.2	AST-8	43.5	2331.2	NN/D-4*	17.4		
AST-B	253.9	7849.1	NN/D-9*	36.0	2367.2	AST-8	17.1	2367.2	15%
NN/D-3*	252.9	8102.0	NN/D-10	34.1	2401.3	NN/D-10	14.6	2401.3	10%
PHY-1C	241.7	8343.7	PHY-3B	26.5	2427.8	PHY-3B	8.7		
NN/D-10	233.6	8577.3	PHY-1C	20.6	2448.4	PHY-1C	8.5		
NN/D-9*	232.6	8809.9	EO-6	10.9	2459.3	AST-1C	7.8		
EO-6	169.5	8979.4	AST-1C	9.5	2468.8	EO-6	6.4		
PHY-1B	126.1	9105.5	PHY-1B	6.7	2475.5	PHY-1B	5.3		
AST-1C	121.3	9226.8	PHY-3A	0	2475.5	PHY-3A	0		

*PROGRAMS INVOLVING DESIGN FAILURES.

SUMMARY

	EXP	OM	Δ	%
LEO	$16,343.6	$ 9,171.6	$6992.2	42.8
MEO & HEO	9,226.8	6,751.3	2475.5	26.8
TOTAL	$25,570.4	$15,922.9	$9467.7	37.0

ALL COST AND SAVINGS ARE MILLIONS OF DOLLARS

best, of the effects on costs of design failures, it does indicate the trend that the ability to take care of design failures does provide an additional cost benefit to the on-orbit maintainable mode. Although not calculated, similar benefits could be shown for the ground refurbishable mode over the expendable mode, although the amount of savings would be smaller.

2. Consideration of Expendable Spacecraft Flights Prior to the Shuttle Era

As discussed previously, and presented in Table IX-5, some programs existed for which expendable spacecraft would be flown prior to 1979 for LEO spacecraft or prior to 1982 for tug-delivered spacecraft. Each program listed in Table IX-5 must make its own decision as far as launching a portion of its spacecraft as expendable plus a portion as on-orbit maintainable. For some programs, the decision is easy. Programs which show a cost penalty for launching in the on-orbit maintainable mode (such as AST-4) should, of course, fly all spacecraft in the expendable mode. Other spacecraft, with only a small percent savings (see Tables IX-33 and IX-34) for flying in the on-orbit maintainable mode (such as PHY-1B, EO-6, AST-1C, PHY-1C) would be strongly advised to also fly in the expendable mode.

Spacecraft which lie in the upper portions of the percent savings list in Tables IX-33 and IX-34 would probably be advised to either delay the expendable launches until the STS is ready, particularly if the period is short and only 1 or 2 expendable spacecraft launches are involved (such as EO-4) or to pay the penalty of developing separate expendable and on-orbit maintainable spacecraft if several missions are to be launched expendably and if the spacecraft program does not want to wait for the STS to be ready for maintenance (possibly NN/D-11). For some programs, the decision may be made easier due to the fact that the early expendable missions may be one model of the spacecraft, and the later missions may have to be a separate model anyway (such as NN/D-1, NN/D-2A and NN/D-2B). This decision must be made individually for each spacecraft, and the suggestions presented here are intended only to list several options, not to attempt to make early decisions for the programs listed.

3. Multiple Spacecraft Servicing

The maintenance costs presented were based upon a single servicing per mission. However, additional savings can be realized from servicing 2 or

more spacecraft per mission. The main savings were from the spreading of the launch costs of the servicer among the sapcecraft to be serviced (module launch costs are already spread by the launch cost reimbursement policy used), although some savings are realized by a slight reduction in maintenance concept costs ($\approx$ \$10-20M).

For tug missions, the additional savings due to multiple spacecraft servicings are estimated to be \$92M for 2 servicings per flight or \$123M for 3 servicings per flight. For LEO spacecraft, the additional savings are estimated to be \$63M, \$84M, and \$94M for 2, 3 or 4 servicings per flight, respectively. The total affects on multiple spacecraft servicings are estimated to be about \$0.2B in addition to the \$9.0B already shown.

4. Expendable Servicer Mechanisms

With the launch cost reimbursement policy used for tug flights, the charge to return a servicer from the spacecraft orbit to the ground is about \$1.2M (including both the tug and orbiter charges).

Table IX-36 presents a summary of the servicer unit costs for a pivoting arm servicer with a 90 percent improvement curve.

Table IX-36 Servicer Unit Costs Learning Curve

NUMBER OF SERVICERS BUILT	AVERAGE COST PER SERVICER
1	\$1.80M
30	\$1.07M
60	\$0.97M
100	\$0.89M

The table shows that if enough servicers are built, it may be cheaper to expend servicer mechanisms than to return them for tug missions. This, of course, is highly dependent upon the final form of launch cost reimbursement policy. However, this does suggest that it may also be economically feasible to consider the use of servicer mechanisms with the Interim Upper Stage (IUS). The mission model used for this study estimated a total of 99 servicing missions requiring the tug (more if design failures are included, less if multiple spacecraft servicing is used). Additional servicing missions flown with the IUS may increase the total savings even more.

K. MAINTENANCE CONCEPT SELECTION

The cost summary has basically addressed, up to now, the primary question of which *mode* - expendable, ground refurbishable, or on-orbit maintainable - provides the most economic benefits during the Shuttle era. However, it is also necessary to address which maintenance *concept* can provide the greatest economic benefits. During this study, up to 25 separate maintenance concepts were evaluated and four were selected to be carried through the study. These four were selected based primarily on technical evaluations. These four, as shown in Table IX-31, were SRMS, EVA, and two servicer mechanisms, the pivoting arm and the general purpose manipulator. The data in Table IX-31, however, does not present the total story. It is necessary to examine the total economic effects of selecting each concept. Two of the concepts (EVA and SRMS) could only be utilized in LEO, and thus would not realize some of the total economic benefits of being able to perform maintenance on the entire range of shuttle automated spacecraft. In addition, the life cycle costs shown in Table IX-31 reflected total maintenance concept costs for the pivoting arm and the general purpose manipulators for performing about 100 more servicings than for the EVA and SRMS. To compare all the concepts more equitably, costs for the same number of missions should be compared. And finally, the selection of any one of the concepts will entail specific spacecraft effects which are not reflected in Table IX-31. Table IX-37 presents another form of

Table IX-37 Visiting System Cost Comparisons (millions of dollars)

MAINTENANCE CONCEPTS	DDT&E	PRODUCTION	OPERATIONS	MAINTENANCE CONCEPT SUBTOTAL	ΔS/C DDT&E AND PRODUCTION EFFECTS	Δ ORBITER LCRP EFFECTS	TOTAL
PIVOTING ARM LEO/MEO/HEO	29	17	57	103	0	0	103
PIVOTING ARM LEO ONLY	29	14	47	90	0	0	90
SRMS, LEO ONLY	22	20	40	82	0	100	182
EVA, LEO ONLY	18	11	51	80	90	100	270
GENERAL PURPOSE MANIPULATOR LEO/MEO/HEO	32	22	64	118	0	40	158

summary of the maintenance concepts where spacecraft program effects are included. The 90 million dollars shown for EVA spacecraft DDT&E and production effects is to show the effects of designing the spacecraft for EVA maintenance. The orbiter LCRP effects for the SRMS and for EVA (100 missions) is to take into account launching and returning the larger spacecraft support structure used for EVA and SRMS maintenance. The orbiter LCRP effects for the general purpose manipulator (40 million) is to take into account launching and returning a heavier servicer to tug orbits. Also shown is the reduced cost for the pivoting arm for flying only LEO missions, to put it on the same basis as the SRMS and EVA. The general purpose manipulator costs for LEO only are not shown since the pivoting arm is shown to be generally a more economic servicer mechanism. Table IX-38 presents a total summary of flying

Table IX-38 Maintenance Modes and Concept Cost Summary

			ON-ORBIT MAINTAINABLE				COMBINATIONS	
	EXPENDABLE	GROUND REFURBISHABLE	EVA	SRMS	GENERAL PURPOSE MANIPULATOR	PIVOTING ARM	LEO-EVA MEO & HEO-GR	LEO-EVA MEO-EVA MEO/HEO-GR
LEO SPACECRAFT	16.34	12.46	9.54	9.45	9.35	9.35	9.54	9.54
MEO & HEO SPACECRAFT	8.53	7.62	8.53	8.53	6.47	6.43	7.62	7.37
MAINTENANCE CONCEPT	-	-	0.08	0.08	0.12	0.10	0.08	0.08
TOTAL	24.87	20.08	18.15	18.06	15.94	15.88	17.24	16.99
Δ FROM EXPENDABLE	0	-4.79	-6.72	-6.81	-8.93	-8.99	-7.63	-7.88
Δ FROM GROUND REFURBISHABLE	+4.79	0	-1.93	-2.02	-4.14	-4.20	-2.84	-3.09

all the programs in the expendable, ground refurbishable, or on-orbit maintainable modes using EVA, SRMS, and the pivoting arm. Various combinations are also shown.

As can be seen from both tables, the pivoting arm can save from 100 to 200 million dollars over EVA and the SRMS, if LEO only is considered, but from 1 to 2 billion dollars, over EVA and the SRMS if the entire shuttle automated spacecraft program is considered. The 9.0 billion dollar savings (shown in Table IX-38) from developing and flying the pivoting arm servicer during the

shuttle era could approach 10 billion or more when other factors are considered (such as design failures, multiple spacecraft servicings, expendable servicer mechanisms, use of the IUS, etc.).

L. COST SENSITIVITY ANALYSIS

The final cost summaries for the IOSS indicated a savings of over 9 billion dollars possible during the shuttle era by developing the capability to perform on-orbit maintenance rather than fly all spacecraft in the expendable mode. A sensitivity study was performed on the data used to calculate total costs and savings to determine:

1) Accuracy and validity of input data and effects of data inputs on results, and

2) Effects of future changes in data on study results.

Results of the cost sensitivity study show that mission model changes and parts factors variations have the greatest effects on study results, but that on-orbit maintenance is still feasible with reductions of up to 50 percent in the number of missions currently in the mission model, for almost all conditions.

The method used to investigate the variation and sensitivity of the cost data is known as the "influence coefficient" method. Let

1) $S = S\ (x_1, x_2, \ldots, x_n)$

where S = savings of on-orbit maintainable over expendable

x_i = cost parameters used to calculate S

2)
$$\frac{\partial S}{\partial x_i} = \frac{\partial}{\partial x_i} S\ (x_1, x_2, \ldots, x_n)$$

where $\dfrac{\partial S}{\partial x_i}$ = influence coefficient for each cost parameter

3)
$$\Delta S = \frac{\partial S}{\partial x_i} \Delta X_i$$

where ΔS = total delta in savings

ΔX_i = delta (or variation) in each cost parameter

Once the influence coefficients have been calculated for all parameters, it is simple to multiply each influence coefficient by any change in any cost parameter to determine what its effect will be on savings. In this analysis, it was determined to only look at savings between expendable and on-orbit maintainable and to ignore ground refurbishable. While ground refurbishable mode may be the preferred mode for some spacecraft, the final cost summary showed that, for the launch cost reimbursement policy used, all spacecraft in the maintenance applicable set were less expensive for either expendable or on-orbit maintainable.

1. Equation for Savings

The following two equations are the basic equations used in the final cost summary for total costs for each spacecraft for expendable and on-orbit maintainable modes:

$$\text{EX cost} = \sum_{i=1}^{n+LF} (a_i C_o + b_i C_t) + (n + LF)(1 + R) C_{S/C} + C_{NR} + C_{KPE}$$

$$\text{OM cost} = \sum_{i=1}^{n+LF} (a_i C_o + b_i C_t)$$

$$+ (n_f + LF_3 + (n - n_f)pf)(1 + R)(1 + sf) C_{S/C} + d\, C_{NR} + C_{KPO}$$

Table IX-3 presented a definition of all parameters used in the costing equations.

Separate equations were made for LEO flights and MEO/HEO flights, and the following substitutions were used, as follows:

a_E = a for expendable flights

b_E = b for expendable flights

a_{01} = a (n_f) for original emplacement launch for on-orbit maintainable flights

a_{02} = a $(n - n_f)$ for servicing launches

b_{01} = b (n_f) for original emplacement launches for on-orbit maintainable flights

b_{02} = b $(n - n_f)$ for servicing launches

The on-orbit maintainable cost was then subtracted from the expendable cost to calculate savings.

$$S_L = C_o \left[n\, a_E - n_f\, a_{01} - (n - n_f)a_{02} + LFa_E - LF\,(n_f)a_{01} - (LF(n - n_f) + LF_1)a_{02} \right]$$

$$+ C_{S/C}\,(1 + R)\left[n + LF - (n_f + LF_3 + (n - n_f)pf)\,(1 + sf)\right]$$

$$+ C_{NR}\,(1 - d)$$

$$+ C_{KPE} - C_{KPO}$$

where S_L = savings equation for LEO

The equation for savings for MEO/HEO ($S_{M/H}$) is the same with the added tug term.

$$C_t\left[n\, b_E - n_f b_{01} - (n - n_f)b_{02} + LFb_E - LF\,(n_f)b_{01} - (LF\,(n - n_f) + LF_1)b_{02}\right]$$

Before the equations were used for the final cost summary, two changes were made. First of all, load factors were used for the orbiter and the tug (to allow for flying orbiters and tugs not fully loaded) as follows:

$$C_o = \frac{C_{12.0}}{\lambda_o} \quad \text{and} \quad C_t = \frac{C_{1.1}}{\lambda_t}$$

where $C_{12.0}$ = actual launch cost of orbiter (\$12.0M was used)

$C_{1.1}$ = actual launch cost of tug (\$1.1M was used)

λ_o = load factor, orbiter (0.70 was used)

λ_t = load factor, tug (0.85 was used)

In addition, launch costs of the on-orbit maintainable mode were altered to take into account the differences between initial emplacement launches for the on-orbit fleet size and servicing launches. Different values of "a" were calculated for the establishment of the fleet size and for servicing launches and the loss factors were altered for the establishment of the

fleet and for the servicing. These resulted in launch costs as follows for on-orbit maintainable.

$$\sum_{i=1}^{n+LF} (a_i C_o + b_i C_t) = (a_{n_f} C_o + b_{n_f} C_t)(n_f + LF(n_f))$$

$$+ (a_{n-n_f} C_o + b_{n-n_f} C_t)(n - n_f + LF(n - n_f) + LF_1(n - n_f))$$

$C_{KPE} - C_{KPO}$ was reduced to

$$C_{MY} \cdot 30 \left(\frac{C_{S/C}}{10}\right)^{.43} \left(\frac{n_f - n}{2}\right)$$

where C_{MY} = cost per man year

A simplifying assumption of $a_{01} \approx a_E$ and $b_{01} \approx b_E$ was used and values for the various "LF's" were used, as shown below:

	LF	LF (n_f)	LF $(n - n_f)$	LF_1 $(n - n_f)$
LEO	.01n	.03 $(n - n_f)$	.01 $(n - n_f)$	$(.06 + 1F_{NRU})(n - n_f)$
MEO/HEO	.02n	.04 $(n - n_f)$	.02 $(n - n_f)$	$(.07 + 1f_{NRU})(n - n_f)$

The LEO and the MEO/HEO equations could then be reduced to the following forms:

$$\begin{aligned} S_L = {} & C_o \left\{ (n - n_f) \left[(a_E - a_o) 1.01 - .03 \right] \right\} \\ & + C_{S/C} (1 + R) \left\{ 1.01\, n - sf \cdot n_f (pf + .06 + 1f_{NRU})(n - n_f) \right\} \\ & - C_{NR} (d - 1) \\ & - C_{MY}\, 30 \left(\frac{C_{S/C}}{10}\right)^{.43} \left(\frac{n - n_f}{2}\right) \end{aligned}$$

$$\begin{aligned} S_{M/H} = {} & C_o \left\{ (n - n_f) \left[(a_E - a_o) 1.02 - .04 \right] \right\} \\ & + C_t \left\{ (n - n_f) \left[(b_E - b_o) 1.02 - .04) \right] \right\} \\ & + C_{S/C} (1 + R) \left\{ 1.02\, n - sf \cdot n_f (pf + .07 + 1f_{NRU})(n - n_f) \right\} \\ & - C_{NR} (d - 1) \\ & - C_{MY} \left(\frac{C_{S/C}}{10}\right)^{.43} \left(\frac{n - n_f}{2}\right) 30 \end{aligned}$$

2. Influence Coefficients

From the savings equations, it can be seen that;

$$S = S\,(n,\ n_f,\ pf,\ sf,\ d,\ R,\ C_{S/C},\ C_{NR},\ C_o,\ C_t,\ (a_E - a_o),\ (b_E - b_o)).$$

It must be remembered that $C_o = \dfrac{C_{12.0}}{\lambda_o}$ and $C_t = \dfrac{C_{1.1}}{\lambda_t}$ and S also can be a a function of $C_{12.0}$, $C_{1.1}$, λ_o and λ_t.

Table IX-39 presents a summary of all the equations used to calculate S_L and $S_{M/H}$ for all the spacecraft. For the influence coefficient for n_f, two separate types were calculated with subscripts 1 and 2. Subscript 1 indicates that n varies with n_f according to the following equation:

$$n_1 = n_f\left(\frac{n}{n_f}\right) \text{ original}$$

Subscript 1 would be used, for an example, on a program like EOS, where the number of operating cycles is held constant at 16, but the on-orbit fleet size is increased from 1 to 4. Subscript 2 would be used, for example in a program like INTELSAT where the originally costed program consisted of a fleet size of 9, each serviced once, which could be cut to a fleet size of 7, each serviced once.

Table IX-40 presents a summary of the nominal values of all the costing parameters for each spacecraft program in the maintenance applicable set and the nominal savings.

Table IX-41 presents a summary of all of the influence coefficients for each spacecraft program in the maintenance applicable set.

The sign of the influence coefficient is very important. A positive sign indicates that a positive delta (increase) of the costing parameter will result in an increase in savings for that program, and a negative delta (decrease) of the costing parameter will result in a decrease in the savings. A negative sign on the influence coefficient will, of course, mean the opposite. For example, an increase in n_f (if n is held constant) would mean a decrease in savings for all spacecraft programs. However, an increase in n_f (if n varies with n_f such that the ratio between the two is held constant) would result in an increase in savings for most programs. Those that show a

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Table IX-39 Influence Coefficient Equations

$$\left(\frac{\partial S_L}{\partial n}\right) = 17.281\left(a_E - a_o\right) - 0.513 + C_{S/C}\left[1.01 - 1.177\left(pf + 0.06 + 1f_{NRU}\right)\right] - 0.326\, C_{S/C}^{\;0.43}$$

$$\left(\frac{\partial S}{\partial n}\right) = 17.452\left(a_E - a_o\right) + 1.316\left(b_E - b_o\right) - 0.736 + C_{S/C}\left[1.112 - 1.177\left(pf + 0.07 + 1f_{NRU}\right)\right] - 0.326\, C_{S/C}^{\;0.43}$$

$$\left(\frac{\partial S_L}{\partial n_f}\right)_1 = -17.281\left(a_E - a_o\right) + 0.513 + 1.177\, C_{S/C}\left(pf + 0.06 + 1f_{NRU}\right) + 0.326\, C_{S/C}^{\;0.43}$$

$$\left(\frac{\partial S}{\partial n_f}\right)_1 = -17.452\left(a_E - a_o\right) - 1.316\left(b_E - b_o\right) + 0.736 + 1.177\, C_{S/C}\left(pf + 0.07 + 1f_{NRU}\right) + 0.326\, C_{S/C}^{\;0.43}$$

$$\left(\frac{\partial S_L}{\partial n_f}\right)_2 = \left(\frac{n}{n_f} - 1\right)\left\{17.281\left(a_E - a_o\right) - 0.513 - C_{S/C}\left[1.177\left(pf + 0.06 + 1f_{NRU}\right)\right] - 0.326\, C_{S/C}^{\;0.43}\right\} + C_{S/C}\left[1.101\frac{n}{n_f} - 1.177\right]$$

$$\left(\frac{\partial S}{\partial n_f}\right)_2 = \left(\frac{n}{n_f} - 1\right)\left\{17.452\left(a_E - a_o\right) + 1.316\left(b_E - b_o\right) - 0.736 - C_{S/C}\left[1.177\left(pf + 0.07 + 1f_{NRU}\right)\right] - 0.326\, C_{S/C}^{\;0.43}\right\} + C_{S/C}\left[1.122\frac{n}{n_f} - 1.177\right]$$

$$\left(\frac{\partial S_L}{\partial p_f}\right) = -1.177\, C_{S/C}\left(n - n_f\right)$$

$$\left(\frac{\partial S}{\partial p_f}\right) = -1.177\, C_{S/C}\left(n - n_f\right)$$

$$\left(\frac{\partial S_L}{\partial s_f}\right) = -1.09\, C_{S/C}\left[n_f + \left(n - n_f\right)\left(pf + 0.06 + 1f_{NRU}\right)\right]$$

$$\left(\frac{\partial S}{\partial s_f}\right) = -1.09\, C_{S/C}\left[n_f + \left(n - n_f\right)\left(pf + 0.07 + 1f_{NRU}\right)\right]$$

$$\left(\frac{\partial S_L}{\partial d}\right) = -C_{NR}$$

$$\left(\frac{\partial S}{\partial d}\right) = -C_{NR}$$

$$\left(\frac{\partial S_L}{\partial R}\right) = -C_{S/C}\left[1.01\, n - 1.08\, n_f - 1.08\left(pf + 0.06 + 1f_{NRU}\right)\left(n - n_f\right)\right]$$

$$\left(\frac{\partial S}{\partial R}\right) = -C_{S/C}\left[1.02\, n - 1.08\, n_f - 1.08\left(pf + 0.07 + 1f_{NRU}\right)\left(n - n_f\right)\right]$$

$$\left(\frac{\partial S_L}{\partial C_{S/C}}\right) = 1.101\, n - 1.177\, n_f - 1.177\left(pf + 0.06 + 1f_{NRU}\right)\left(n - n_f\right) - \frac{0.139\left(n - n_f\right)}{C_{S/C}^{\;0.57}}$$

$$\left(\frac{\partial S}{\partial C_{S/C}}\right) = 1.112 - 1.177\, n_f - 1.177\left(pf + 0.07 + 1f_{NRU}\right)\left(n - n_f\right) - \frac{0.139\left(n - n_f\right)}{C_{S/C}^{\;0.57}}$$

$$\left(\frac{\partial S_L}{\partial C_{NR}}\right) = -\left(d - 1\right)$$

$$\left(\frac{\partial S}{\partial C_{NR}}\right) = -\left(d - 1\right)$$

$$\left(\frac{\partial S_L}{\partial C_o}\right) = \left(n - n_f\right)\left|\left(a_E - a_o\right) 1.01 - 0.03\right|$$

$$\left(\frac{\partial S}{\partial C_o}\right) = \left(n - n_f\right)\left|\left(a_E - a_o\right) 1.02 - 0.04\right|$$

$$\left(\frac{\partial S_L}{\partial C_{S/C}}\right) = 1.101\, n - 1.177\, n_f - 1.177 \left(pf + 0.06 + 1f_{NRU}\right)\left(n - n_f\right) - \frac{\cdots}{C_{S/C}{}^{0.57}}$$

$$\left(\frac{\partial S}{\partial C_{S/C}}\right) = 1.112 - 1.177\, n_f - 1.177 \left(pf + 0.07 + 1f_{NRU}\right)\left(n - n_f\right) - \frac{0.139 \left(n - n_f\right)}{C_{S/C}{}^{0.57}}$$

$$\left(\frac{\partial S_L}{\partial C_{NR}}\right) = -(d - 1)$$

$$\left(\frac{\partial S}{\partial C_{NR}}\right) = -(d - 1)$$

$$\left(\frac{\partial S_L}{\partial C_0}\right) = \left(n - n_f\right)\left[\left(a_E - a_0\right) 1.01 - 0.03\right]$$

$$\left(\frac{\partial S}{\partial C_0}\right) = \left(n - n_f\right)\left[\left(a_E - a_0\right) 1.02 - 0.04\right]$$

$$\left(\frac{\partial S}{\partial C_t}\right) = \left(n - n_f\right)\left[\left(b_E - b_0\right) 1.02 - 0.04\right]$$

$$\left(\frac{\partial S_L}{\partial a_E - a_0}\right) = 17.281 \left(n - n_f\right)$$

$$\left(\frac{\partial S}{\partial a_E - a_0}\right) = 17.452 \left(n - n_f\right)$$

$$\left(\frac{\partial S}{\partial b_E - b_0}\right) = 1.316 \left(n - n_f\right)$$

$$\left(\frac{\partial S_L}{\partial C_{12.0}}\right) = \frac{\left(n - n_f\right)}{\lambda_0}\left[\left(a_E - a_0\right) 1.01 - 0.03\right]$$

$$\left(\frac{\partial S}{\partial C_{12.0}}\right) = \frac{\left(n - n_f\right)}{\lambda_0}\left[\left(a_E - a_0\right) 1.02 - 0.04\right]$$

$$\left(\frac{\partial S}{C_{1.1}}\right) = \frac{\left(n - n_f\right)}{\lambda_t}\left[\left(b_E - b_0\right) 1.02 - 0.04\right]$$

$$\left(\frac{\partial S_L}{\partial \lambda_0}\right) = -\frac{12.0}{\lambda_0^2} \left(n - n_f\right)\left[\left(a_E - a_0\right) 1.01 - 0.03\right]$$

$$\left(\frac{\partial S}{\partial \lambda_0}\right) = -\frac{12.0}{\lambda_0^2} \left(n - n_f\right)\left[\left(a_E - a_0\right) 1.02 - 0.04\right]$$

$$\left(\frac{\partial S}{\partial \lambda_t}\right) = -\frac{1.1}{\lambda_t^2} \left(n - n_f\right)\left[\left(b_E - b_0\right) 1.02 - 0.04\right]$$

S_L = SAVINGS IN LEO

$S = S_{M/H}$ = SAVINGS IN MEO/HEO

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Table IX-40 Nominal Values of Cost Parameters

Payload No.	Payload Model Code No.	LEO Spacecraft Name	a_E-a_O	pf	$1f_{NRU}$	b_E-b_O	n	n_f	sf
AS-03-A	AST-1B	Cosmic Background Explorer	.13	.25	.01		7	1	.08
SO-03-A	AST-3	Solar Maximum Mission	.17	.17	.01		6	1	
HE-09-A	AST-4	Large High Energy Observatory B	.21	.09	.03		1	1	
HE-03-A	AST-5A	Extended X-Ray Survey	.21	.09	.03		3	1	
HE-08-A	AST-5B	Large High Energy Observatory A	.21	.09	.03		3	1	
HE-10-A	AST-5C	Large High Energy Observatory C	.21	.09	.03		2	1	
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	.21	.09	.03		1	1	
AS-01-A	AST-6	Large Space Telescope	.39	.09	.03		12	1	
SO-02-A	AST-7	Large Solar Observatory	.43	.09	.03		7	1	
HE-11-A	AST-9A	Large High Energy Observatory D	.30	.09	.03		4	1	
HE-01-A	AST-9B	Large X-Ray Telescope Facility	.49	.06	.03		3	1	
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	.30	.09	.03		9	1	
AS-11-A	AST-N2	1.5m IR Telescope	.21	.09	.03		11	1	
AS-13-A	AST-N3	UV Survey Telescope	.21	.09	.03		6	1	
AS-14-A	AST-N4	1m UV - Optical Telescope	.21	.09	.03		11	1	
AS-17-A	AST-N5	30m IR Interferometer	.55	.09	.03		4	1	
HE-07-A	PHY-1A	Small High Energy Satellite	.12	.09	.01		6	1	
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	.19	.12	.01		2	1	
HE-12-A	PHY-5	Cosmic Ray Laboratory	.21	.09	.03		5	1	
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	.27	.10	.01		25	1	
EO-08-A	EO-3	Earth Observatory Satellite	.36	.16	.01		16	2	
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	.10	.29	.01		16	2	
OP-02-A	EOP-5	Gravity Gradiometer	.17	.09	.01		1	1	
OP-04-A	EOP-7	GRAVSAT	.10	.24	.01		1	1	
OP-05-A	EOP-8	Vector Magnetometer Satellite	.11	.28	.01		9	3	
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	.19	.10	.01		9	3	
		MEO/HEO Spacecraft Name							
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	.33	.22	.01	.21	3	1	.08
AS-16-A	AST-8	Large Radio Observatory Array	0	.19	.02	0	4	1	
AP-01-A	PHY-1B	Upper Atmosphere Explorer	.01	.09	.01	0	2	1	
AP-02-A	PHY-1C	Explorer-Medium Altitude	0	.08	.01	0	3	1	
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	.02	.21	.01	0	1	1	
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	.12	.15	.01	0	2	1	
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	.42	.15	.02	.33	8	2	
EO-12-A	EO-6	TIROS	.06	.19	.01	0	2	1	
CN-51-A	NN/D-1	INTELSAT	.21	.28	.02	.11	18	9	
CN-53-A	NN/D-2B	DOMSAT B	.35	.28	.02	.25	14	7	
CN-58-A	NN/D-2C	DOMSAT C	.22	.31	.02	.10	6	3	
CN-54-A	NN/D-3	Disaster Warning Satellite	.19	.24	.02	.08	4	2	
CN-55-A	NN/D-4	Traffic Management Satellite	.09	.29	.02	.01	14	7	
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	.11	.24	.02	.01	24	12	
CN-59-A	NN/D-6	Communications R&D Prototype	.25	.25	.02	.15	3	1	
EO-56-A	NN/D-8	Environmental Monitoring Satellite	.05	.16	.01	0	7	1	
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	.08	.37	.01	-.02	6	2	
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	.08	.38	.01	-.02	8	2	
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	.03	.13	.01	0	11	2	
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	.42	.15	.02	.33	10	2	
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	.42	.14	.02	.33	10	2	

* In millions of dollars

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	a_E-a_O	pf	$1f_{NRU}$	b_E-b_O	n	n_f	sf	d	R	λ_o	λ_T	C_O*	C_T*	$C_{12.0}$*	$C_{1.1}$*	Savings*
	.13	.25	.01		7	1	.08	.04	.09	0.70		17.1		12.0		39.5
	.17	.17	.01		6	1										167.6
	.21	.09	.03		1	1										-11.5
	.21	.09	.03		3	1										134.4
	.21	.09	.03		3	1										141.7
	.21	.09	.03		2	1										44.9
	.21	.09	.03		1	1										-13.7
	.39	.09	.03		12	1										976.1
	.43	.09	.03		7	1										524.6
	.30	.09	.03		4	1										207.2
	.49	.06	.03		3	1										171.5
	.30	.09	.03		9	1										565.7
	.21	.09	.03		11	1										550.3
	.21	.09	.03		6	1										214.3
	.21	.09	.03		11	1										445.4
	.55	.09	.03		4	1										141.7
	.12	.09	.01		6	1										330.3
	.19	.12	.01		2	1										16.7
	.21	.09	.03		5	1										452.4
	.27	.10	.01		25	1										528.5
	.36	.16	.01		16	2										1052.6
e)	.10	.29	.01		16	2										60.6
	.17	.09	.01		1	1										-10.5
	.10	.24	.01		1	1										-9.0
	.11	.28	.01		9	3										39.2
	.19	.10	.01		9	3										187.0

	a_E-a_O	pf	$1f_{NRU}$	b_E-b_O	n	n_f	sf	d	R	λ_o	λ_T	C_O*	C_T*	$C_{12.0}$*	$C_{1.1}$*	Savings*
	.33	.22	.01	.21	3	1	.08	.04	.09	0.70	0.85	17.1	1.3	12.0	1.1	9.5
	0	.19	.02	0	4	1										43.5
	.01	.09	.01	0	2	1										6.7
	0	.08	.01	0	3	1										20.6
	.02	.21	.01	0	1	1										-11.3
	.12	.15	.01	0	2	1										26.5
	.42	.15	.02	.33	8	2										191.3
	.06	.19	.01	0	2	1										10.9
	.21	.28	.02	.11	18	9										160.0
	.35	.28	.02	.25	14	7										130.4
	.22	.31	.02	.10	6	3										19.7
	.19	.24	.02	.08	4	2										20.6
	.09	.29	.02	.01	14	7										36.3
	.11	.24	.02	.01	24	12										61.3
	.25	.25	.02	.15	3	1										38.7
	.05	.16	.01	0	7	1										107.5
	.08	.37	.01	-.02	6	2										20.0
llite	.08	.38	.01	-.02	8	2										34.1
	.03	.13	.01	0	11	2										638.4
	.42	.15	.02	.33	10	2										256.5
e	.42	.14	.02	.33	10	2										266.1

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Table IX-41 Summary of Influence Coefficients

Payload No.	Payload Model Code No.	LEO Spacecraft Name	$\left(\frac{ds}{dn}\right)$	$\left(\frac{ds}{dn_{f1}}\right)$	$\left(\frac{ds}{dn_{f2}}\right)$	$\left(\frac{ds}{dpf}\right)$	$\left(\frac{ds}{dsf}\right)$	$\left(\frac{ds}{dd}\right)$	$\left(\frac{ds}{dR}\right)$
AS-03-A	AST-1B	Cosmic Background Explorer	6.527	-7.112	38.576	-54.377	-24.508	-47.7	30.156
SO-03-A	AST-3	Solar Maximum Mission	35.016	-38.193	171.902	-245.993	-100.238	-166.2	153.991
HE-09-A	AST-4	Large High Energy Observatory B	57.196	-61.983	-4.788	0	-68.67	-168.3	-4.41
HE-03-A	AST-5A	Extended X-Ray Survey	74.208	-80.470	142.153	-193.970	-122.150	-220.3	128.643
HE-08-A	AST-5B	Large High Energy Observatory A	78.51	-84.117	150.385	-205.504	-118.728	-233.4	136.293
HE-10-A	AST-5C	Large High Energy Observatory C	55.619	-60.270	50.968	-72.032	-78.715	-163.5	45.631
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey	67.978	-73.700	-5.723	0	-82.077	-201.2	-5.271
AS-01-A	AST-6	Large Space Telescope	89.879	-97.227	981.311	-1251.975	-314.101	-258.5	860.785
SO-02-A	AST-7	Large Solar Observatory	89.955	-97.230	532.430	-677.952	-217.651	-263.3	463.066
HE-11-A	AST-9A	Large High Energy Observatory D	73.832	-79.927	215.401	-283.186	-134.624	-224.5	190.619
HE-01-A	AST-9B	Large X-Ray Telescope Facility	94.016	-101.281	180.765	-225.042	-135.465	-255.4	149.251
AS-07-A	AST-N1	3m Ambient Temperature IR Telescope	72.077	-78.020	570.673	-736.331	-207.981	-208.8	504.765
AS-11-A	AST-N2	1.5m IR Telescope	55.707	-60.365	552.407	-721.501	-187.088	-163.8	495.672
AS-13-A	AST-N3	UV Survey Telescope	44.160	-47.815	217.142	-283.068	-99.615	-128.4	192.785
AS-14-A	AST-N4	1m UV - Optical Telescope	44.946	-48.670	445.735	-576.73	-149.548	-131.1	396.21
AS-17-A	AST-N5	30m IR Interferometer	50.21	-53.880	146.957	-170.547	-81.077	-129.1	114.799
HE-07-A	PHY-1A	Small High Energy Satellite	67.384	-73.039	331.267	-437.844	-145.973	-77.1	306.230
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO	22.720	-24.559	20.880	-28.483	-31.390	-116.8	17.782
HE-12-A	PHY-5	Cosmic Ray Laboratory	117.909	-127.999	461.597	-621.927	-247.661	-271.5	421.716
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite	21.865	-23.461	523.153	-593.208	-116.281	-147.6	415.036
EO-08-A	EO-3	Earth Observatory Satellite	76.538	-83.226	529.078	-1450.064	-500.702	-284.8	925.971
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)	4.012	-4.400	27.698	-84.038	-39.135	-53.0	43.640
OP-02-A	EOP-5	Gravity Gradiometer	49.269	-53.319	-4.051	0	-58.097	-157.5	-3.731
OP-04-A	EOP-7	GRAVSAT	29.212	-32.267	-3.055	0	-43.818	-160.2	-2.814
OP-05-A	EOP-8	Vector Magnetometer Satellite	6.881	-7.580	16.448	-64.970	-51.143	-68.3	32.954
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System	32.975	-35.650	63.274	-248.582	-154.239	-136.9	167.144
		MEO/HEO Spacecraft Name							
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer	10.789	-11.328	21.287	-19.538	-14.475	-51.7	11.056
AS-16-A	AST-8	Large Radio Observatory Array	16.619	-17.912	48.565	-82.978	-47.132	-79.6	49.181
AP-01-A	PHY-1B	Upper Atmosphere Explorer	10.047	-10.871	9.476	-14.948	-16.196	-68.3	9.860
AP-02-A	PHY-1C	Explorer-Medium Altitude	13.327	-14.393	26.080	-38.606	-23.596	-138.1	26.804
AP-05-A	PHY-3A	Environmental Perturbation Satellite-A	34.807	-37.921	-2.635	0	-52.211	-195.4	-2.874
AP-07-A	PHY-3B	Environmental Perturbation Satellite-B	36.632	-39.487	34.658	-51.670	-58.857	-169.1	31.239
EO-09-A	EO-4	Synchronous Earth Observatory Satellite	34.720	-37.008	103.279	-248.582	-131.986	-136.4	156.457
EO-12-A	EO-6	TIROS	15.639	-16.971	15.452	-24.128	-28.155	-98.6	13.924
CN-51-A	NN/D-1	INTELSAT	20.912	-22.758	19.634	-300.841	-381.687	-116.0	143.238
CN-53-A	NN/D-2B	DOMSAT B	23.605	-25.458	22.323	-234.812	-297.913	-117.3	111.800
CN-58-A	NN/D-2C	DOMSAT C	10.832	-11.703	10.229	-47.315	-61.345	-73.8	21.226
CN-54-A	NN/D-3	Disaster Warning Satellite	14.719	-15.902	13.900	-42.843	-52.769	-94.6	21.971
CN-55-A	NN/D-4	Traffic Management Satellite	6.932	-7.614	6.459	-86.510	-110.559	-72.0	40.396
CN-56-A	NN/D-5A	Foreign Communication Satellite-A	8.105	-8.807	7.619	-152.539	-187.881	-73.7	78.227
CN-59-A	NN/D-6	Communications R&D Prototype	24.810	-26.857	48.517	-74.151	-57.683	-141.5	39.236
EO-56-A	NN/D-8	Environmental Monitoring Satellite	18.325	-20.152	111.750	-168.076	-63.298	-107.5	107.214
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite	7.112	-7.944	13.776	-60.262	-53.018	-81.9	25.805
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite	7.112	-7.944	20.565	-90.394	-66.412	-81.9	38.646
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite	73.145	-78.026	328.278	-925.828	-370.585	-281.1	613.443
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite	39.882	-37.183	138,997	-333.326	-151.257	-137.6	211.211
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite	35.710	-38.044	142.302	-338.034	-150.263	-140.5	217.296

All coefficients are in millions of 1975 dollars per unit change in parameter except for those marked * where the coefficients are dimensionless.

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$\left(\frac{ds}{dn}\right)$	$\left(\frac{ds}{dn_{f1}}\right)$	$\left(\frac{ds}{dn_{f2}}\right)$	$\left(\frac{ds}{dpf}\right)$	$\left(\frac{ds}{dsf}\right)$	$\left(\frac{ds}{dd}\right)$	$\left(\frac{ds}{dR}\right)$	$\left(\frac{ds}{d\lambda_0}\right)$	$\left(\frac{ds}{d\lambda_T}\right)$	$\left(\frac{ds}{d(a_E-a_0)}\right)$	$\left(\frac{ds}{d(b_E-b_0)}\right)$	$\left(\frac{ds}{dC_{S/C}}\right)^*$	$\left(\frac{ds}{dC_{NR}}\right)^*$	$\left(\frac{ds}{dC_0}\right)^*$	$\left(\frac{ds}{dC_T}\right)^*$	$\left(\frac{ds}{dC_{12.0}}\right)^*$	$\left(\frac{ds}{dC_{1.1}}\right)^*$
6.527	-7.112	38.576	-54.377	-24.508	-47.7	30.156	-7.294		103.69		4.010	-.04	.608		.869	
35.016	-38.193	171.902	-245.993	-100.238	-166.2	153.991	-8.502		86.41		3.934	-.04	.709		1.013	
57.196	-61.983	-4.788	0	-68.67	-168.3	-4.41	0		-		-.076	-.04	0		0	
74.208	-80.470	142.153	-193.970	-122.150	-220.3	128.643	-4.370		34.56		1.680	-.04	.364		.52	
78.51	-84.117	150.385	-205.504	-118.728	-233.4	136.293	-4.370		34.56		1.681	-.04	.364		.52	
55.619	-60.270	50.968	-72.032	-78.715	-163.5	45.631	-2.185		17.28		.800	-.04	.182		.26	
67.978	-73.700	-5.723	0	-82.077	-201.2	-5.271	0		-		-.076	-.04	0		0	
89.879	-97.227	981.311	-1251.975	-314.101	-258.5	860.785	-48.035		190.09		9.592	-.04	4.008		5.719	
39.955	-97.230	532.430	-677.952	-217.651	-263.3	463.066	-29.110		103.69		5.197	-.04	2.426		3.466	
73.832	-79.927	215.401	-283.186	-134.624	-224.5	190.619	-9.828		51.84		2.557	-.04	.819		1.170	
94.016	-101.281	180.765	-225.042	-135.465	-255.4	149.251	-11.158		34.56		1.752	-.04	.930		1.329	
72.077	-78.020	570.673	-736.331	-207.981	-208.8	504.765	-26.208		138.25		6.944	-.04	2.184		3.120	
55.707	-60.365	552.407	-721.501	-187.088	-163.8	495.672	-21.852		172.81		8.682	-.04	1.821		2.601	
44.160	-47.815	217.142	-283.068	-99.615	-128.4	192.785	-10.926		86.41		4.293	-.04	.910		1.300	
44.946	-48.670	445.735	-576.73	-149.548	-131.1	396.21	-21.852		172.81		8.664	-.04	1.821		2.601	
50.21	-53.880	146.957	-170.547	-81.077	-129.1	114.799	-18.918		51.84		2.546	-.04	1.576		2.251	
67.384	-73.039	331.267	-437.844	-145.973	-77.1	306.230	-5.472		86.41		4.428	-.04	.456		.651	
22.720	-24.559	20.880	-28.483	-31.390	-116.8	17.782	-1.943		17.28		.779	-.04	.162		.231	
17.909	-127.999	461.597	-621.927	-247.661	-271.5	421.716	-8.741		69.12		3.446	-.04	.728		.040	
21.865	-23.461	523.153	-593.208	-116.281	-147.6	415.036	-69.898		414.74		20.958	-.04	5.825		8.321	
76.538	-83.226	529.078	-1450.064	-500.702	-284.8	925.971	-56.045		241.93		11.320	-.04	4.670		6.671	
4.012	-4.400	27.698	-84.038	-39.135	-53.0	43.640	-11.928		241.93		8.561	-.04	.952		1.36	
49.269	-53.319	-4.051	0	-58.097	-157.5	-3.731	0		-		-.076	-.04	0		0	
29.212	-32.267	-3.055	0	-43.818	-160.2	-2.814	0		-		-.076	-.04	0		0	
6.881	-7.580	16.448	-64.970	-51.143	-68.3	32.954	-5.839		103.69		3.671	-.04	.487		.696	
32.975	-35.650	63.274	-248.582	-154.239	-136.9	167.144	-11.657		103.69		5.068	-.04	.971		1.387	
10.789	-11.328	21.287	-19.538	-14.475	-51.7	11.056	-7.118	-.383	34.90	2.63	1.370	-.04	.593	3.48	.847	.409
16.619	-17.912	48.565	-82.978	-47.132	-79.6	49.181	+1.44	+.132	52.36	3.95	2.213	-.04	-.120	-.120	-.170	-.141
10.047	-10.871	9.476	-14.948	-16.196	-68.3	9.860	+.358	+.044	17.45	1.32	.814	-.04	-.030	-.040	-.043	-.047
13.327	-14.393	26.080	-38.606	-23.596	-138.1	26.804	+.96	+.088	34.90	2.63	1.726	-.04	-.08	-.080	-.114	-.094
34.807	-37.921	-2.635	0	-52.211	-195.4	-2.874	0	0	-	-	-.076	-.04	0	0	0	0
36.632	-39.487	34.658	-51.670	-58.857	-169.1	31.239	-.989	+.088	17.45	1.32	.760	-.04	.082	-.080	.117	-.094
34.720	-37.008	103.279	-248.582	-131.986	-136.4	156.457	-27.965	-1.958	104.71	7.90	4.824	-.04	2.330	1.78	3.329	2.094
15.639	-16.971	15.452	-24.128	-28.155	-98.6	13.924	-.254	+.088	17.45	1.32	.716	-.04	.021	-.080	.03	-.094
20.912	-22.758	19.634	-300.841	-381.687	-116.0	143.238	-18.814	-.715	157.17	11.84	5.318	-.04	1.568	.65	2.24	.765
23.605	-25.458	22.323	-234.812	-297.913	-117.3	111.800	-26.628	-1.656	122.16	9.21	4.136	-.04	2.219	1.505	3.170	1.771
10.832	-11.703	10.229	-47.315	-61.345	-73.8	21.226	-6.638	-.205	52.36	3.95	1.634	-.04	.553	.186	.790	.219
14.719	-15.902	13.900	-42.843	-52.769	-94.6	21.971	-3.691	-.092	34.90	2.63	1.264	-.04	.308	.084	.440	.099
6.932	-7.614	6.459	-86.510	-110.559	-72.0	40.396	-4.351	+.229	122.16	9.21	4.083	-.04	.363	-.208	.519	-.245
8.105	-8.807	7.619	-152.539	-187.881	-73.7	78.227	-10.397	+.393	209.42	15.79	7.713	-.04	.866	-.357	1.237	-.42
24.810	-26.857	48.517	-74.151	-57.683	-141.5	39.236	-5.16	-.249	34.90	2.63	1.320	-.04	.430	.226	.614	.266
18.325	-20.152	111.750	-168.076	-63.298	-107.5	107.214	-.792	+.264	104.71	7.90	4.775	-.04	.066	-.24	.094	-.282
7.112	-7.944	13.776	-60.262	-53.018	-81.9	25.805	-1.997	+.266	69.81	5.26	2.069	-.04	.166	-.242	.237	-.285
7.112	-7.944	20.565	-90.394	-66.412	-81.9	38.646	-2.995	+.399	104.71	7.90	3.098	-.04	.250	-.363	.357	-.427
73.145	-78.026	328.278	-925.828	-370.585	-281.1	613.443	+1.015	+.396	157.07	11.84	7.556	-.04	-.085	-.36	-.121	-.424
39.882	-37.183	138.997	-333.326	-151.257	-137.6	211.211	-37.286	-2.610	139.62	10.53	6.461	-.04	3.107	2.372	4.439	2.791
35.710	-38.044	142.302	-338.034	-150.263	-140.5	217.296	-37.286	-2.610	139.62	10.53	6.456	-.04	3.107	2.372	4.439	2.791

rameter except for

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decrease in savings are programs with only one operating cycle. Table IX-42 presents a summary of which cost parameters will result in increases or decreases with positive deltas.

Table IX-42 Summary of Cost Parameter Deltas Sign Effects

COST PARAMETERS WHICH INCREASE SAVINGS FOR POSITIVE DELTA	COST PARAMETERS WHICH DECREASE SAVINGS FOR POSITIVE DELTA
n	n_{f_1}
n_{f_2}*	pf
R*	sf
$(a_E - a_0)$	d
$(b_E - b_0)$	λ_0
$C_{S/C}$*	λ_T
C_0	
C_T	C_{NR}
$C_{12.0}$	
$C_{1.1}$	
*WILL DECREASE SAVINGS IF ONLY 1 FLIGHT PER PROGRAM.	

Once the influence coefficients were determined, the next step was to reexamine the cost parameters to determine the possible variations. The main effect on the costs and savings comes from possible changes in the mission model used to cost expendable and maintainable programs.

At this date, the spacecraft programs that will be flying on the shuttle are very nebulous and it is necessary to know what possible changes will do to on-orbit maintenance. Any changes in the mission model will affect mainly the parameters n and n_f, although the end effects of mission model changes may result in changes to parameters associated with the launch cost reimbursement policy, λ_o, λ_t, C_o, C_t, a, b, etc. For this analysis, only the changes in n and n_f were examined for changes in the mission model. Changes in the LCRP were looked at separately.

The mission model used to cost spacecraft programs for this study is the maintenance applicable set, and was discussed in Chapter III.

It consists of 47 spacecraft programs with 317 missions from 1979 through 1991 and an additional 23 missions after 1991. Only the years of mission launches are required, it was not necessary to assign mission to shuttle flights due to the method of calculating LCRP. If correctly loaded on board shuttles and tugs, the LCRP indicates that the entire maintenance applicable set could be flown using 168 shuttle flights and 84 tug flights, if all missions are flown expendably or 100 shuttle flights and 84 tug flights, if flown in the on-orbit maintainable mode. (This assumed a load factor of 0.70 for the orbiter and 0.85 for the tug.) The current traffic model lists a total of 782 shuttle flights, with 540 flights being available for NASA and commercial missions. (The October 1973 Space Shuttle Traffic Model, NASA TM X-64751, Rev. 2, MSFC, January 1974, Bibliography Item A-2.) Other mission models have suggested a reduction in the total number of flights to 572 or to 546. (The "572 Flight Shuttle Traffic Model" and "Generation of Shuttle Flights Manifests for the October 1974 Traffic Model", NASA Memorandum, D. N. Turner, 10 June 1975, Bibliography Item A-6.) These represent, roughly, a reduction in the total number of flights by about 25 percent.

Since the costs were determined for the maintenances applicable set independent of which specific flights the missions were flown on, and since it appears that the specific flight data may vary as the shuttle era draws closer, it was decided to vary the mission model used for costing maintenance independently of specific flight data. Two separate cases were investigated; a reduction in the mission model of 25 percent of the operating cycles (missions) and of 50 percent in the operating cycles. No programs were eliminated in the reduction of operating cycles. Operating cycles were reduced according to the following rules:

1) Programs with 1 or 2 cycles - no reductions.

2) Programs with 3 cycles - reduce 1.

3) Programs with 4 or more cycles reduce at least 25% (or 50%) to have each program and total number close to 25% (or 50%) reduction. This applied to programs with $n_f = 1$.

4) For programs with large n_f, reduce n_f (according to the #2 method, where $n = n\ (n_f)$ using rule 3) for n_f instead of n.

The actual number of operating cycles was reduced from 340 to 249 for a 25 percent reduction and from 340 to 171 for a 50 percent reduction. Results of this showed that total savings were reduced from 9 billion dollars (for 340 cycles) to 5.9 billion dollars for a 25 percent reduction and 3.3 billion dollars for a 50 percent reduction.

The deviation in parts factor (pf), factor to modify S/C unit cost (sf), factor to modify DDT&E (d), and ratio of launch checkout to S/C cost (R), for the on-orbit maintainable mode were calculated as follows:

$$D = D_1 + D_2 + D_3$$

where D = total deviation in parts factor values

D_1 = deviation due to inaccuracies in data sample

D_2 = deviation about the mean of the data sample

D_3 = deviation of the mean due to the limited size of the data sample

$$D_1 = \frac{\Delta E}{\sqrt{n}} = \frac{\text{Individual inaccuracies}}{\sqrt{\text{number of data samples}}}$$

$$D_2 = \sqrt{\frac{\sum(x-\bar{x})^2}{n-1}} = \text{ = standard deviation about the mean}$$

$$D_3 = \frac{\sigma}{\sqrt{n}} \sqrt{1 - \frac{n}{N}} =$$ deviation of the mean due to limited data sample. If n = N (the data sample represents all of the possible cases) then $D_3 = 0$. If $\frac{n}{N} \approx 0$ (the data sample is very limited) then $D_3 = \frac{\sigma}{\sqrt{n}}$

3. Sensitivity Analysis Results

Table IX-43 presents a summary of data and results for variations in pf, sf, d, and R. Results indicate that for an average parts factor of about 0.16 a variation of a max of ± 0.08 is reasonable, for a value of sf of 0.08 a variation of a max of $\pm .06$ is reasonable, for a value of d of 0.04 a variation of a max of ± 0.03 is reasonable and for a value of R of 0.09, a variation of $\pm .03$ is reasonable.

Applying the correct influence coefficient for each spacecraft program for each of these cost parameters, the total cost parameters can be calculated. Results of these calculations are presented in Table IX-44. Note that increases in R tend to increase savings while increases in pf, sf, and d all

Table IX-43 Summary of Deviation in pf, sf, d, and R

COST PARAMETER	pf	sf	d	R
n	25	6	4	3
$\bar{x}$	0.16	0.08	0.04	0.09
ΔE	±0.045	±0.02	±0.01	±0.0225
D_1	±0.0106	±0.0082	±0.0050	±0.0075
D_2	±0.0589	±0.0312	±0.0124	±0.0115
N	47	?	?	?
n/N	0.53	0	0	0
D_3	±0.0080	±0.0127	±0.0062	±0.0067
D	±0.0776	±0.0521	±0.0237	±0.0257
D_{USED}	±0.08	±0.06	±0.03	±0.03

Table IX-44 Summary of Cost Variations for pf, sf, d, and R

COST PARAMETER	pf	sf	d	R
DEVIATION	±0.08	±0.06	±0.03	±0.03
ΔS_{LEO} ($M)	∓738.5	∓210.7	∓133.1	±185.3
$\Delta S_{MEO/HEO}$ ($M)	∓266.8	∓142.6	∓73.7	±59.0
ΔS_{TOTAL} ($M)	∓1005.3	∓353.3	∓206.8	±244.3

tend to decrease savings. Also note that the variation in costs due to pf is larger than all the others. It is also possible to sum the Δ's in saving and to root sum square (RSS) these deltas in savings, which is probably more correct since all are pretty much independent. The sum of the absolute magnitude of these values is 1.8 billion and the RSS value is 1.1 billion, both fairly small as compared to the 9 billion or more total savings expected from on-orbit maintenance.

The same type of analyses as applied to pf, sf, d and R was applied to the cost parameters assocated with the LCRP. Table IX-45 presents a summary to the data used to calculate deviations for these parameters. Table IX-46 then presents the summary of Δ's in savings due to applying the cost parameters variations in Table IX-46 to the individual influence coefficients for each of the 47 spacecraft programs. Again, the individual Δ's can be summed and RRS'd to result in values of $^{+1454.5}_{-1117.6}$ for the sum and $^{+890.4}_{-719.0}$ for the RSS.

It would be possible to combine the Δ's calculated for the LCRP along with the remaining maintenance cost parameters, but it was decided not to because: (1) it was believed that the LCRP parameters were a lot more nebulous than the other parameters and 2) while the variations in most of the maintenance cost parameters could easily be in either directions (+ or -), the expected change for the LCRP parameters are expected to be mostly in a known direction (+ for $C_{12.0}$ and $C_{1.1}$ and - for λ_o and λ_t). In fact, a separate calculation was made for orbiter launch costs. If the cost of launching the orbiter were to increase from $12.0M per flight to $20M per flight, the savings obtained by flying in the on-orbit maintainable mode would be increased from $9.0B to $9.6B (the increase in savings would be $556M).

Figures IX-17 and IX-18 present plots of total savings in going from expendable spacecraft programs to on-orbit maintainable spacecraft programs as a function of what percentage of the maintenance applicable set is flown. Plotted in Figure IX-17 are the possible variations in cost due to maximum expected variations in pf, sf, d and R and plotted on Figure IX-18 are possible variations in cost due to maximum expected variations in the LCRP parameters. For both Figures IX-17 and IX-18 it can be seen that for reductions in the mission model of up to 50 percent and for a wide range

Table IX-45 Summary of Deviations for LCRP Parameters

COST PARAMETER	(\$M) $C_{12.0}$	(\$M) $C_{1.1}$	λ_{ORB}	λ_{TUG}	$(a_E - a_o)$	$(b_E - b_o)$
n	--	--	--	--	47	21
$\bar{x}$	12.0	1.1	0.70	0.85	0.207	0.089
ΔE	--	--	--	--	± 0.10	± 0.10
D_1	--	--	--	--	± 0.0146	± 0.0218
D_2	--	--	--	--	+0.1358	+0.126
N	--	--	--	--	47	21
n/N	--	--	--	--	1	1
D_3	--	--	--	--	0	0
D	--	--	--	--	± 0.1504	± 0.1478
D_{USED}	± 3.0	± 0.3	± 0.3	± 0.15	± 0.15	± 0.15

Table IX-46 Summary of Cost Variations for LCRP Parameters

COST PARAMETER	$C_{12.0}$	$C_{1.1}$	λ_{ORB}	λ_{TUG}	$(a_E - a_o)$	$(b_E - b_o)$
DEVIATION	± 3.0	± 0.3	± 0.3	± 0.15	± 0.15	± 0.15
ΔS_{LEO} (\$M)	± 141.3	--	-164.5 +392.4	--	± 383.6	--
$\Delta S_{MEO/HEO}$ (\$M)	± 67.4	± 2.6	-78.3 +187.0	-1.3 +1.6	± 259.2	± 19.5
ΔS_{TOTAL} (\$M)	± 208.7	± 2.6	-242.8 +579.4	-1.3 +1.6	± 642.8	± 19.5

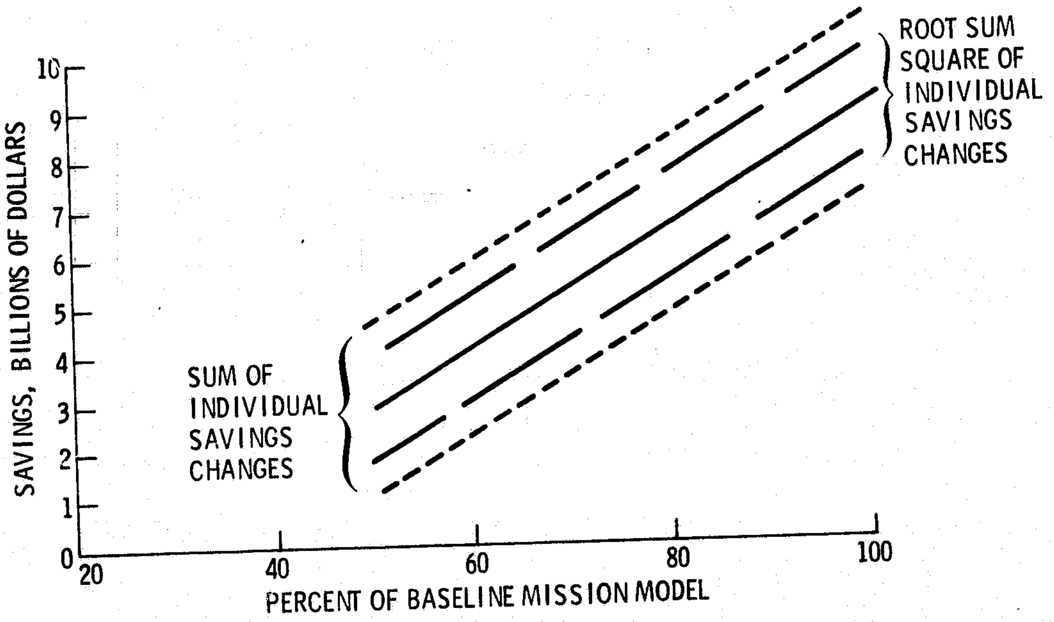


Figure IX-17 Cost Sensitivity Analysis, Partial Summary for pf, sf, d, R

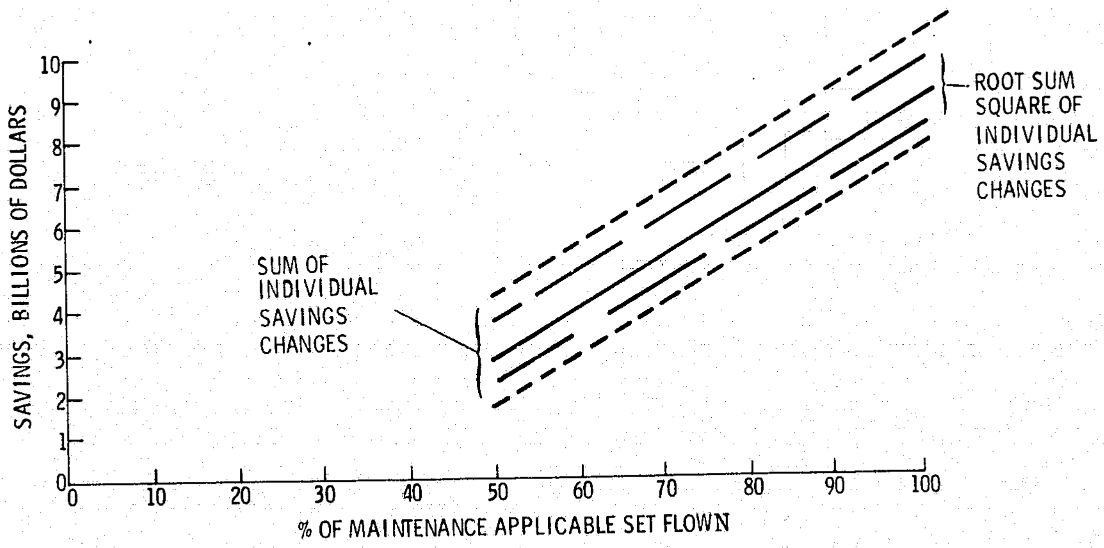


Figure IX-18 Cost Sensitivity Analysis, Partial Summary for LCRP

of possible variations in cost parameters, on-orbit servicing can still save billions of dollars over expendable spacecraft programs.

Many other variations in the input data are possible and, in fact, will come about long before the shuttle era. The data provided in this report will enable any interested party to calculate for himself the effects on the total savings (or on individual spacecraft savings) due to any changes in any of the input data by using the influence coefficients and nominal savings given in Table IX-41.

M. COST ANALYSIS SUMMARY

The important conclusions that can be drawn from the economic evaluation include:

1) Use of on-orbit servicing over the 12 years covered by the 1974 SSPD and the October 1973 Payload Model results in savings greater than
   - nine billion dollars over the expendable mode, and
   - four billion dollars over the ground refurbishable mode.
2) The life cycle costs of the on-orbit servicer represent approximately one percent of the overall savings and these costs can be fully recovered by 1982.
3) Cost sensitivity analyses showed that wide variations in cost data, especially mission model size and fraction of spacecraft replaced, affect specific savings but do not change the major study conclusions.
4) A long-life free-flying servicer at geostationary orbit is potentially cost effective.
5) Specific launch cost reimbursement policies can be an important factor in which form of servicing is adopted for individual spacecraft programs.
6) Expendable satellites are cost effective where satellite lifetime meets program lifetime requirements.

X. SELECTION OF A RECOMMENDED SYSTEM

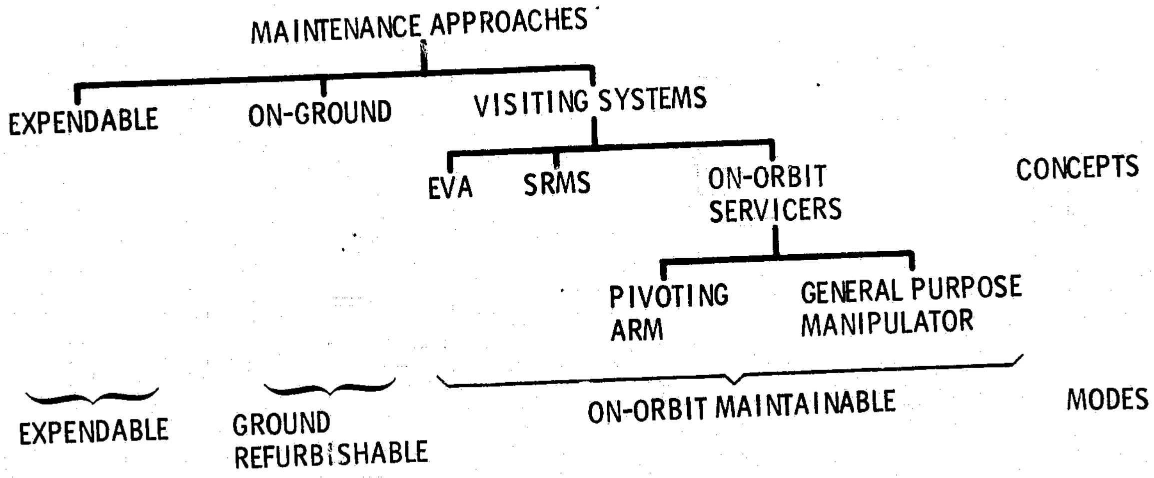
Chapters III through IX have developed material and presented evaluations, tradeoffs, and selections in the various aspects of orbital maintenance. However, each of the chapters addresses part of the story; this chapter brings it all together to select the final recommended maintenance concept. All of the necessary data was developed above; the important factors will now be isolated and brought together in a logical way. The selection of a cost effective orbital maintenance system and the supporting rationale are the primary study outputs.

A. SELECTION APPROACH

The approach to the final selection logic is shown in Figure X-1. The three maintenance modes--expendable, ground refurbishable, and on-orbit maintainable--are shown along with the alternative maintenance concepts for each mode that survived the prior evaluations and tradeoffs. The order of conducting this final selection will be from the bottom up. The first selection will be between the on-orbit servicers--pivoting arm and general-purpose manipulator--,the second between the visiting systems--EVA, the shuttle remote manipulator system, and the selected on-orbit servicer--,and the third between the three maintenance modes.

The evaluation considerations listed are those that were used throughout the study and those that were used in this final selection. Spacecraft design aspects involved whether spacecraft can be designed to be compatible with a given maintenance mode with acceptable design, weight, volume, and cost effects. This has been expressed as minimum constraints on the spacecraft design. The spacecraft program mission requirements and schedules should not be compromised by spacecraft design for servicing. The STS impacts involve the effects of the maintenance concept on the STS and include such things as rendezvous and docking, weight capability to desired orbits, communications requirements, and integration into the ground processing flow.

Technical considerations involve whether a maintenance concept is technically feasible and involve such things as weight, stowed length, design requirements, docking system compatibility, reliability, simplicity, operability, and need for advances in the state-of-the-art. Operational



EVALUATION CONSIDERATIONS:

1. SPACECRAFT DESIGN ASPECTS
2. STS IMPACTS
3. TECHNICAL
4. OPERATIONAL AREAS
5. PROGRAMMATIC
6. COST

Figure X-1 Maintenance Modes and Concepts

areas involve application of a maintenance concept to low-, medium-, and high-earth orbits.

The programmatic considerations involve such things as whether more than one maintenance concept must be developed, whether the development schedule is compatible with the need, growth capability, and the number of spacecraft programs where the maintenance concept is applicable. There are two levels of cost considerations. Maintenance concept costs involve DDT&E, production, and operations for the maintenance concept only. Spacecraft program costs involve the spacecraft DDT&E, production, and operations costs as well as the launch costs associated with the spacecraft and the maintenance concept.

The spaceborne part of the maintenance concepts involved in the final selection are shown in Fig. X-2. The expendable and ground refurbishable concepts are also involved in the final selection.

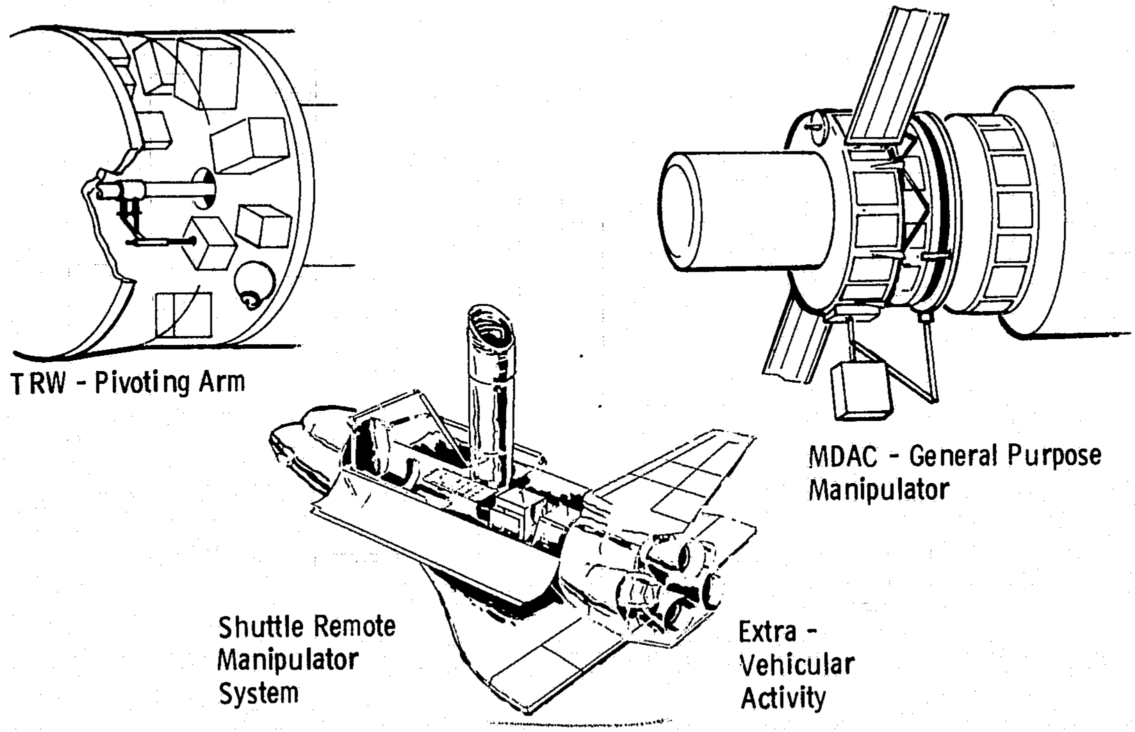


Figure X-2 Remaining Visiting Systems

B. ON-ORBIT SERVICER SELECTION

The two maintenance concepts involved in this level of selection are the pivoting arm and the general purpose manipulator. Each has been configured to do basically the same module exchange activity. They have similar spacecraft design effects with the important difference being that the pivoting arm primarily replaces modules axially, while the general purpose manipulator primarily replaces modules radially. However, either can be developed to handle modules in both directions. The pivoting arm places fewer constraints on the spacecraft designer and because it uses the entire end of the spacecraft it permits a slightly better spacecraft volumetric efficiency (the central region is not lost).

Either concept is adaptable to central or peripheral docking mechanisms. They both satisfy all the design criteria and are technically feasible. The pivoting arm is lighter and simpler, while the general purpose manipulator has a shorter stowed length. The general purpose manipulator is 15 percent higher in maintenance concept cost, but when the

difference in total spacecraft program costs are also considered, the general purpose manipulator costs can be as much as 50 percent higher. Both can be applied at all spacecraft orbits, but the general purpose manipulator is more adaptable to complex servicing tasks should they be required.

The results of this evaluation are summarized in Table X-1. The totality of the factors favors the pivoting arm concept and it is selected and recommended. The preliminary design of this concept was accomplished successfully (Chapter VII) which strengthens this recommendation.

Table X-1 On-Orbit Servicer Selection Summary

RECOMMEND PIVOTING ARM
LEVEL OF AVAILABLE SPACECRAFT DEFINITION HAS NOT PERMITTED IDENTIFICATION OF A SPECIFIC REQUIREMENT FOR RADIAL MODULE REPLACEMENT TO THIS POINT.
VERSATILITY FOR RADIAL MODULE REPLACEMENT SHOULD BE OBTAINED THROUGH CONCEPT GROWTH.
LITTLE DIFFERENCE IN STS IMPACT
SIMPLEST
NO DIFFERENCE IN OPERATIONAL AREAS
SIMILAR DEVELOPMENT PLANS AND OPERATIONAL CONSIDERATIONS
LEAST COST

C. VISITING SYSTEM COMPARISON

The three maintenance concepts involved in this level of selection are (1) EVA, (2) shuttle remote manipulator system, and (3) pivoting arm servicer. They are all evaluated on their ability to perform module exchange, although each does it in different ways and with different effects. The EVA and SRMS concepts are limited to operations at the orbiter. An early cost analysis showed that the cost of using two tugs and orbiters to bring a high earth orbit spacecraft to the orbiter for maintenance and then to replace it in its orbit was excessive. Thus this option is not further

considered. Only single tug/orbiter options are considered. The result is that the majority of this evaluation involves operations at the orbiter. The effect of this limitation is brought into the cost discussion.

The three maintenance concepts have been defined and evaluated in Chapter IV. All have similar spacecraft design effects in that each concept has been defined to perform axial module exchange using similar module sizes, SRU interface mechanisms, and end effectors (power tools for EVA). EVA places some additional design conditions on the spacecraft. These are not major but do inovlve such things as sharp edges, restraint attachments, delicate surface protection, and hand grip sizes for EVA gloves. These effects on spacecraft design will show up as increased spacecraft cost. The effect will be small and is estimated to be one percent of spacecraft non-recurring and recurring cost.

All the three concepts are compatible with central or peripheral docking devices. The SRMS is used to dock spacecraft for each of the concepts and to move concept equipment (e.g., stowage rack) from launch/reentry location to and from the servicing operation location in the orbiter cargo bay. Each of the three concepts are technically feasible. EVA requires development of a power tool for module replacement which has been included in the EVA costs. The current SRMS design requirements do not include module exchange, rather they emphasize payload deployment and retrieval. The addition of a module exchange capability to the SRMS will result in increments in development, production, and operation costs. These cost increments have been developed and discussed in Chapter IX and will be shown in the maintenance concept costs below. The pivoting arm design can be essentially the same for use on the orbiter, tug, EOTS, a geosynchronous free-flyer, or some forms of the IUS. Each maintenance concept can use the same form of module stowage rack.

The airborne support equipment (ASE) for the three concepts is different. The pivoting arm concept is easily placed into an operating position by the SRMS. It contains its own spacecraft docking mechanism, has a short stowage length and simple, foldable, support structure between the stowage rack and the orbiter when in the operating position.

Both EVA and SRMS require a large bulky structure to support large spacecraft, with deployed appendages, above the orbiter mould line and

clear of the orbiter tail. This structure must also span from orbiter longeron to longeron. It takes the form of a truss work that is approximately 17 ft x 17 ft x 10 ft when in the operating configuration. It must be folded to get it into the cargo bay (doors closed) and further folded to decrease its stowed length. The technique for this folding has not been addressed, but is possible. A stowed length of five feet is estimated which results in an increment in the launch costs for SRMS and EVA over the pivoting arm for large spacecraft. Small spacecraft with acceptable appendages can be placed in the orbiter cargo bay for servicing and present no launch cost penalties.

Several additional considerations are shown in Table X-2 which complete the non-cost considerations. The non-cost considerations may be summarized as:

(1) Equivalent spacecraft design effects;

(2) Equivalent STS impacts, except for orbiter cargo bay stowage aspects;

(3) Each is technically feasible;

(4) Only the pivoting arm can be used in high earth orbit;

(5) Programmatic considerations --

a) SRMS operations are more awkward,

b) EVA and SRMS use more STS baseline equipment,

c) Use of pivoting arm in LEO implies a single visiting system development.

Visiting system maintenance concept life-cycle costs are shown in Table X-3 in millions of dollars. The EVA and SRMS costs shown are those associated with servicing and which are in addition to the STS baseline costs. The pivoting arm costs are shown for LEO/MEO/HEO which represent a capability to service all 47 spacecraft programs. They are also shown when the capability is reduced to LEO only so that they may be compared with the SRMS and EVA costs. Note that for LEO only, the direct costs of the different visiting systems are estimated to only vary by ±6 percent (from 80 to 90 million dollars). However, when the additional spacecraft costs for EVA,

Table X-2 Additional Considerations for Visiting Systems

SERVICING TIME

EACH CONCEPT REQUIRES 1 TO 2.5 HOUR RANGE WHICH IS NOT A SIGNIFICANT DIFFERENCE. HOWEVER, EVA REQUIRES AN ADDITIONAL 5 HOURS FOR PRE- AND POST-EVA ACTIVITIES.

CARGO BAY OPERATIONS

EVA AND PIVOTING ARM OPERATING LOCATIONS ARE PRIMARILY LIMITED BY SPACECRAFT APPENDAGES. SRMS IS ALSO LIMITED BY ITS OWN REACH AND CONFIGURATION EFFECTS.

UTILIZATION OF EXISTING STS BASELINE EQUIPMENT

EVA AND SRMS USE MORE OF THE STS BASELINE EQUIPMENT THAN THE ON-ORBIT SERVICER.

SERVICING ORBITS

EVA AND SRMS ARE LIMITED TO LOW EARTH ORBITS WHILE THE ON-ORBIT SERVICER CAN BE UTILIZED IN HIGH EARTH ORBITS ALSO.

DEVELOPMENT OF THE ON-ORBIT SERVICER FOR LEO APPLICATIONS

BENEFITS THE EXTENSION OF SERVICING TO HEO APPLICATIONS

ESTABLISHES USER ACCEPTANCE BEFORE HEO SPACECRAFT DESIGNS BECOME FIXED.

Table X-3 Visiting System Cost Comparisons (millions of dollars)

MAINTENANCE CONCEPTS	DDT&E	PRODUCTION	OPERATIONS	MAINTENANCE CONCEPT SUBTOTAL	Δ S/C DDT&E AND PRODUCTION EFFECTS	Δ ORBITER LCRP EFFECTS	TOTAL
PIVOTING ARM LEO/MEO/HEO	29	17	57	103	0	0	103
PIVOTING ARM LEO ONLY	29	14	47	90	0	0	90
SRMS, LEO ONLY	22	20	40	82	0	100	182
EVA, LEO ONLY	18	11	51	80	90	100	270

and the incremental launch cost reimbursement policy effects for EVA and SRMS are included, the on-orbit servicer could be some 90 million dollars cheaper than the SRMS and some 180 million dollars cheaper than EVA.

The total spacecraft program costs for the three visiting systems are shown in Table X-4. When looked at across the total mission model, including MEO/HEO orbits, the pivoting arm servicer could save an additional 2.2

Table X-4 Visiting Systems - Spacecraft Program Costs (Billions of Dollars)

	PIVOTING ARM	EVA (1)	SRMS (1)	SRMS (2)
LEO SPACECRAFT	9.35	9.54	9.45	9.45
MEO AND HEO SPACECRAFT	6.43	8.53	8.53	7.37
MAINTENANCE CONCEPTS	0.10	0.08	0.08	0.08
TOTAL	15.88	18.15	18.06	16.90

(1) MEO AND HEO FLIGHTS ARE IN EXPENDABLE MODE

(2) BEST MIX OF SRMS, GROUND REFURBISHABLE AND EXPENDABLE

billion dollars over the EVA and SRMS concepts when the MEO and HEO flights are conducted in an expendable mode. The fourth column shows a best mix using SRMS at the orbiter for LEO, a single tug returning some spacecraft to the orbiter for SRMS servicing, and then replacing the spacecraft in orbit for some MEO, and the least expensive of expendable or ground-refurbishable for the other MEO and HEO spacecraft. The pivoting arm servicer provides one billion dollars additional savings when compared to the best of the other visiting system alternatives.

The pivoting arm form of on-orbit servicer is thus selected to represent the on-orbit maintainable mode and, as will be justified below, is thus recommended for further development.

D. MAINTENANCE MODE COMPARISON

The three maintenance modes involved at this level of selection are the expendable, ground refurbishable, and on-orbit maintainable, where the pivoting arm servicer is the concept used for the on-orbit maintainable

mode. The expendable mode just uses expendable spacecraft and thus may be considered to be represented by a valid concept. The ground refurbishable mode only required some changes in the spacecraft for rendezvous, docking, and retrieval and thus also may be considered to be represented by a valid maintenance concept. Table X-5 summarizes the non-cost considerations.

Table X-5 Maintenance Mode Non-Cost Considerations

SPACECRAFT CAN BE DESIGNED FOR EACH MODE - EFFECT IS REFLECTED IN COSTS
STS REQUIREMENTS ARE PART OF STS BASELINE OR EFFECTS ARE REFLECTED IN COSTS
EACH OF THE THREE MODES IS TECHNICALLY FEASIBLE
EACH OF THE THREE MODES CAN BE USED IN LEO, MEO AND HEO
ACCEPTABLE DEVELOPMENT PROGRAMS AND OPERATIONAL METHODS ARE POSSIBLE FOR EACH MODE - EFFECTS ARE REFLECTED IN COSTS

Each of the three modes is feasible and practical. All non-equivalent aspects can be, and have been, expressed as costs.

Table X-6 presents a cost summary of flying the 47 spacecraft programs

Table X-6 Maintenance Mode Cost Summary (Billions of Dollars)

	EXPENDABLE	GROUND REFURBISHABLE	ON-ORBIT MAINTAINABLE
LEO SPACECRAFT	16.3	12.5	9.3
MEO/HEO	8.5	7.6	6.4
MAINTENANCE CONCEPT	--	--	0.1
TOTAL	24.9	20.1	15.9
	--	4.8	9.0

and 340 missions of the mission model in each of the three modes. In the first column, all missions are flown expendably and the total life cycle cost for all the programs is 25 billion dollars. In the second column, most missions are flown in the ground refurbishable mode, although some are still flown in an expendable mode (those that would be cheaper to fly in an expendable mode), and the total life cycle cost is about 20

billion dollars, a savings of almost 5 billion dollars. The third column represents most programs being flown in the on-orbit maintainable mode (as typified by the pivoting arm servicer). Some programs are still cheaper to fly in an expendable mode, but none are cheaper in the ground refurbishable mode. The total cost is less than 16 billion dollars and that represents a savings of some nine billion dollars, or 36 percent, over flying all spacecraft expendably and some 4.2 billion dollars, or 21 percent over the ground refurbishable mode.

The total life cycle costs for the pivoting arm servicer of 103 million dollars represents approximately 1 percent of the nine billion dollars that can be saved by utilizing on-orbit maintenance instead of all expendable spacecraft. As noted in the sensitivity analysis of chapter IX, expected variations in cost elements change the value of savings, but in all cases, the on-orbit maintainable mode is the least expensive.

It is concluded that on-orbit maintenance is the most cost effective mode.

One of the cost sensitivity factors investigated was the size of the mission model. It was shown that reductions in the size of the mission model reduced the size of the savings. It also is recognized that if a spacecraft program is initiated in an expendable mode and then later switched to the more cost effective on-orbit maintainable mode, then the cost of the second development will reduce the possible savings. Both of these factors imply that early acceptance of on-orbit maintenance should be actively pursued. This in turn implies that development of the pivoting arm on-orbit servicer should include early demonstrations to strengthen user acceptance. It also implies that analysis, design, engineering test unit fabrication and evaluation of on-orbit servicers should be continued actively.

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